

# HISTORY OF GERMAN GUIDED MISSILES DEVELOPMENT

AGARD

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## FOREWORD

The information presented in this volume represents the first effort by AGARD to disseminate information on the rapidly growing field of guided missiles.

The subject was chosen because it covers a complete development program. There are many lessons to be learned from the study of such a complete program, the difficulties encountered, and the methods by which these difficulties were overcome.

THEODORE VON KÁRMÁN  
Chairman, AGARD

## EDITORS' PREFACE

During the Fifth General Assembly of AGARD at Ottawa, in June 1955, Dr. VON KÁRMÁN suggested that, as an introduction to possible activity in guided missiles, the Germans be invited to arrange a meeting at which their work on guided missiles developed prior to 1945 should be described.

In Germany, comprehensive research and development of this subject had been started. Numerous missiles had been constructed, and various types were developed ready for active service and were built in large numbers. The considerable experience in research and development of these missiles was, at the end of the war, taken over by several countries, but only a few scientists and engineers are likely to have had an insight into the whole of the German research work, the more so as there were but a few publications on this subject. Thus it appeared advisable to report on this extensive work and then to publish it.

Almost all scientists and engineers who took a leading part in these developments accepted the invitation to this meeting, which took place at Munich from the 23rd to 27th of April 1956. Discussions following the lectures yielded additional information and suggestions. On the whole, an impressive survey was given of the extensive research and development work carried out in Germany. The present AGARDograph contains the papers of the Munich Symposium.

The scientific programme was composed by the two National Delegates of the German Federal Republic to AGARD, who were specially assisted by Prof. Dr.-Ing. O. LUTZ (DEUTSCHE FORSCHUNGSANSTALT FÜR LUFTFAHRT (DFL), Braunschweig) and by Dipl.-Ing. F. MÜNSTER (DEUTSCHE VERSUCHSANSTALT FÜR LUFTFAHRT (DVL), Mülheim/Ruhr).

The WISSENSCHAFTLICHE GESELLSCHAFT FÜR LUFTFAHRT (WGL) was charged with the outside organization of the Seminar. The WGL was also responsible for the editorial work connected with this book, in which Oberregierungsrat Dipl.-Ing. M. MAYER (DEUTSCHES PATENTAMT, Munich) and Dr. E. W. C. WILKINS (recently of the Royal Aeronautical Society, London, now with the Lockheed Aircraft Corporation in California) and his assistants checked the extensive English translations.

Special thanks are due to all persons connected with the organization for the good success of the meeting and of this book.

Bonn and Aachen, June 1957

Dr. TH. BENECKE	Prof. Dr.-Ing. A. W. QUICK
National Delegates of the German Federal Republic to AGARD	

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## SUMMARY OF GERMAN DEVELOPMENTS IN GUIDED MISSILES

THEODOR BENECKE \*

Some years ago when, for the first time after the war, a scientific meeting was held in the German Federal Republic, to discuss rockets, a Swiss rocket expert commenced his lecture with the words: "I need not say anything about the history of rockets in front of German experts, because the history of modern rockets has been written in Germany."

In giving an abstract of German guided missile development I take the liberty of quoting these words, since this history is closely connected with the development of the modern rocket.

One task of the AGARD Congress is to describe the history of German developments in guided missiles, before and during the war, and to consider them as a basis and example for discussing the principal problems of remote-controlled missiles. For the first time we have a meeting of all "guided-missiles experts" of nations united within the NORTH ATLANTIC TREATY ORGANIZATION, and as I see it, the purpose of this Congress is to provide the human contact which is most essential and necessary for the improvement of our joint work.

As the lectures held today, and to be held on the following days, will once more make us thoroughly familiar with many technical details of the various systems, I should like first to consider a question which is common to all the remote-controlled missiles, viz.: Why was it that just these weapons were, in those days, developed in Germany?

Remote-controlled missiles owe their origin to the techniques of two formerly separate fields, i. e. projectiles and aeroplanes. Rocket-driven projectiles have been used for military purposes for several hundred years, but the guided missile did not become possible until advanced aerodynamics, automatic guidance and high-frequency techniques were applied to the modern rocket. The origin of the individual German weapons is revealed by their appearance: the V-2 was derived from the projectile, whereas the V-1 and Hs 293 were derived from the aeroplane.

In addition to these a third type of guided missile, but one which was free of any form of drive, was developed at the beginning of the war, i. e. the guided, free-falling bomb, an example of which was the SD 1400 X.

According to the present scheme of classification, the above four weapons, all of which were used in action, fall into the following categories, viz.:

surface-to-surface, V-2, V-1;

air-to-surface, SD 1400 X (FRITZ X), Hs 293.

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Before returning to the question of why Germany produced these developments, we must take into account the fact that neither the tactical use nor the military demands for these new weapons existed, because the technical possibilities, and their limits, at those times could not be recognised. The tactical problems which could be solved with these new weapons had to be derived from the slowly progressing development. It is understandable that, under these circumstances and the pressure of the situation, serious divergences often arose between the engineers and soldiers. It is understandable that the development only advanced step by step, because its realization was at the limit of any technical possibilities. Furthermore, new testing methods for proving had to be developed, and there were already so many units required for tests that these had to be produced on a large scale, since they were nearly always lost on proving. Under these conditions the development could only have been brought to a successful conclusion if all the persons concerned, and entrusted, with either research work, development, proving, production problems, or the tactical use, had co-operated closely and resigned from any personal concerns in favour of the task in question.

The development of the A-4 (or V-2) derived from an emergency — which often is such a good instructor — i. e. the restrictions on long-range artillery in Germany after the first World War. The work leading to the development of the V-2 was begun in the year 1929, in order to create a mobile, effective weapon — instead of the prohibited medium artillery — to strengthen the fighting power of the small army. The Army Ordnance Department (HEERES-WAFFENAMT) commenced its own rocket development in 1932 in Kummersdorf under the leadership of General DORNBERGER and Professor VON BRAUN. Looking back, one can say that the year 1932 was of decisive importance in the overall development of rockets, and therefore in the field of guided missiles. The taking up of work on liquid-fuel rocket motors meant that, for the first time in history, an important organization had set itself to produce a guided missile which could be used in action against point targets. General DORNBERGER has submitted a unique document about the history of the V-2 in his publication "The Shot into the Universe".

Now I would like to recall just a few dates.

The V-2 was a self-steering, liquid-fuel rocket whose trajectory became purely ballistic after the propellant cut-off. On October 3, 1942, the first successful shot reached a height of 90,000 m, higher than any previous attempt, even with the aid of a gun. The maximum speed in 45° flight, after a vertical start, was 1500 m/sec, i. e. five times the speed of sound. With this launching, the door was opened to a new, worldwide field of engineering.

The military use of the V-2 commenced on September 8, 1944. The first explosive loaded guided missile was launched, and the bombers handed over their duties to the new artillery. All in all, 1,115 of the V-2 were launched against England, and 2,050 against Brussels, Antwerp, and Liège.

The question of whether, when looked at rationally from the point of view of her resources, Germany was right to carry out the extensive development of the V-2 during the war, cannot be discussed here. One thing is certain however, and that is that the V-2 was the basis on which the large-scale post-war developments, especially in the United States, were established.



*Fig. 1. Survey on German guided missiles*

*(From "Nauticus", January 1953, Tables 19 and 20. Reproduced by the courtesy of the Publishers, E. S. Mittler & Son, Darmstadt, Germany)*

The second remote-controlled weapon in the "surface-to-surface" category which was in service during the war, was the Fi-103 or the V-1. The proposal to develop an aeroplane-like device, with a pulse-jet as drive, was submitted at the beginning of the war by the Air Ministry Technical Office (TECHNISCHES AMT of the REICHSLUFTFAHRTMINISTERIUM). The proposal, however, was rejected by the General Staff on the grounds that it could only be used against extended targets, and therefore unavoidably against the civilian population.

Only after the regrettable commencement of the air raids on open cities was the project reconsidered under the code word — remarkable for its purpose —

"KIRSCHKERN", because a cherry stone can be re-expectorated, and now it was executed in an astonishingly short time.

Military operations began on June 13, 1944, with a stock of 5,000 pieces, so that development, testing, troop training and the beginning of production were carried out in two years and three days, from June 10, 1942. This success, with which the names of PAUL SCHMIDT, GOSSLAU, LUSSE and BRÉE are connected, was only made possible by first-class co-operation between the various groups involved.

The development of the individual construction groups for the V-1, such as the catapult equipment, the driving aggregate, the automatic steering and the equipment for determining the flight time, was put into operation at the same time, according to a fixed plan as described in the original proposal, and the production engineers were early employed.

The operations with this flying bomb, measuring 8 m in length and weighing 2.2 tons, of which 800 kg were explosive, began shortly after the invasion, on June 13, 1944. Out of a total of 8,000 V-1 missiles launched against the London area, 2,000 were lost immediately, or shortly after the start, and of the remaining 6,000 missiles, 2,400 went through the concentrated anti-aircraft firing and reached the target. Next to the V-2, the V-1 was the most important solution of the "ground-to-ground" problem developed during the war.

Two successful weapons in the "air-to-surface" category which were used in action during the war were the

SD 1400 X or FRITZ X (see Fig. 2 in Prof. WAGNER's paper on page 9 of this book)

and the Hs 293 (see Fig. 4 in Prof. WAGNER's paper on page 11).

Both were initiated by the Air Force (LUFTWAFFE), to enable the bomb-aimer to correct, during the fall of the bomb, the errors caused at release, and by the wind, and also to meet the general desire to be able to drop a bomb without having to fly over the target.

In 1938, Dr. MAX KRAMER began his experiments in the DVL in Berlin-Adlershof, on affecting the trajectory of a 250 kg bomb by means of wireless control from an aeroplane. For controlling the vertical and lateral movements, and for stabilisation about the roll axis, he developed new steering devices which took the form of movable spoilers attached to a cruciform tail. The normal release procedure was used, but after release its position was determined by a target-covering method. This meant that, after launching, the carrier plane had to maintain its direction of flight, and that the bomb-aimer had to guide the bomb first in a forwards, but later in a rearwards, direction.

In 1940, the armour piercing bomb SD 1400 was chosen as a warhead for this weapon proposed mainly against armoured warships, and test missiles with this bomb were built. The designation of the missile was, therefore, SD 1400 X (code name FRITZ X), the "X" being derived from the cruciform tail. During the first provings, although the terminal velocity of the bomb was limited by a brake-ring to 280 m/sec, difficulties were encountered with the steering effectiveness of the spoilers. To speed up the testing programme which, because of weather conditions and a minimum dropping height of 4,000 m, was faced with considerable difficulties within German territory, the testing was carried out in Foggia, Italy, during March/April 1942. Here the extensive proving

programme was completed in less than four weeks, and it was possible to carry out the large scale production of the SD 1400 X. In addition, the difficulties with the spoilers were eliminated, largely by means of tests in the recently completed high speed wind tunnel at the DVL. During the tests 50 % of the hits released from heights of between 4,000 and 7,000 m were within a 5 m square.

The first action of the SD 1400 X against warships, which was immediately successful, took place in the Mediterranean on August 29, 1943, four days after the first employment at the front of the other guided missile, the Hs 293.

Whilst the project for non-remote-controlled glider bombs, the development of which was always fascinating, and therefore repeatedly taken up again, was deleted from the programme by the Technical Controller's Staff shortly after the beginning of the war, it was decided at the end of 1939 that the remote-controlled glider bomb Hs 293, as projected by Professor HERBERT WAGNER, should be developed. The first remote-controlled test drop took place on December 17, 1940, less than twelve months after the order to take up the development had been given.

Whereas the SD 1400 X was developed for use against armoured warships, the Hs 293, consisting of an SC 500 (500 kg high explosive bomb), was used for the destruction of unprotected or lightly armoured vessels. A similar target covering method was used for directing the remote control. The Hs 293 was similar in construction to a small aeroplane, and it was powered by a short-duration WALTER rocket motor, which was used, after release, to bring the bomb on to its trajectory between the aeroplane and the target. Both the ailerons and the elevator were operated by remote control.

The 50 % target hitting precision, from a dropping distance of up to 12 km, was roughly similar to that of FRITZ X. The bomb could be released, at distances between 3.5 and 18 km, from heights of between 400 and 2,000 m, with a maximum speed of 200 m/sec at the propellant cut-off.

The first time that the Hs 293 was taken into action was in the Biskaya on August 25, 1943. On this occasion, and also later, mostly destroyers were sunk by the Hs 293.

All in all, the target hitting results obtained with both remote-controlled bombs amounted to approximately 40 % of all. It may be surprising that the wave-lengths used for guidance of this missile were just those reserved for this field at the International Conference of Cairo in 1936. As it was possible that the transmission of remote-control orders within the ultra-short wave-range would soon be jammed, a wire guidance method was developed for both the SD 1400 X and the Hs 293. It was produced on a large scale, so that any of the methods could be used, but the wire-control system was not used in operations since no interference was encountered. Incontestable drops were made with wire-remote-control during proving, at distances of up to 18 km. During the last months of the war, the Hs 293 was fired not only against ships but also against bridge installations.

In addition to the weapons described above, all of which were used operationally during the war, a large number of other developments were taken up and partly completed without, however, being used in action. I should now like to give a short account of some of these.

The fact that the destruction of the German industrial plants by bombing became more and more severe as the war progressed, made the development of the surface-to-air rocket and of the air-to-air missile a matter of the utmost importance. This priority was the main reason for the high standard of development reached, in the various forms of both weapons, by the end of the war.

There were four anti-aircraft rockets, known as: WASSERFALL, RHEINTOCHTER, ENZIAN and SCHMETTERLING. The maximum speeds of WASSERFALL and RHEINTOCHTER were in the supersonic region, but ENZIAN and SCHMETTERLING had subsonic maximum speeds.

The development of WASSERFALL was begun at Peenemünde in 1942, along similar lines to the V-2. It was designed to destroy massed bomber squadrons, and to combat individual high flying aircraft, but the problems to be solved were partly well beyond the scope of a V-2 development. Therefore we were forced to use remote-control procedures based on automatic homing devices, because although at that time the beam-control technique was the most advanced, its accuracy decreased with increasing range.

In order to obtain a favourable effect of the explosive on the target, it was necessary to develop proximity-fuse systems which ignited automatically at the minimum distance from the target. So that the rocket could follow pursuit curves at high speed, it was fitted with four stub wings, attached to the body in a symmetrical, cruciform pattern.

The aerodynamic tests were made in the supersonic wind tunnel in Peenemünde, and in this connection I may recall that at the time very little data on supersonic aeroplanes were available.

The remote-control system, which was based on the beam-control principle, had two radar sets, one aimed constantly at the target, the other at the missile. The orders necessary to keep the rocket on the guide beam were transmitted to the steering motors in the rocket by means of a control stick arrangement on the ground.

This work on the prototype remote-controlled anti-aircraft rocket, of which approximately 35 trial units had been launched up to the beginning of 1945, had to be terminated in February 1945 because of displacement of the research plant from Peenemünde to the middle of Germany.

The experience gained in Germany with the WASSERFALL development had its outcome in the well known NIKE development in the United States.

In the same way as the development of the WASSERFALL was derived from the older V-2, the other anti-aircraft rocket SCHMETTERLING had its origin in the experience gathered from the development of the Hs 293, and at the end of the war SCHMETTERLING was the most developed anti-aircraft rocket.

The construction of the Hs 117 SCHMETTERLING was similar in all respects to that of a modern anti-aircraft rocket, and the extremely low all-up-weight of 450 kg is remarkable. It could be fired from a stand, at various angles, but in order to reduce the flight time to a minimum and yet still be able to hit targets at close range, the usual procedure was to fire oblique shots. In this procedure, the SCHMETTERLING was started from a point support, i.e. a zero-length launcher, with the assistance of two powder booster rockets. The main power unit, consisting of a BMW liquid fuel motor, was switched on at the same time as the boosters and, after releasing the boosters, accelerated the SCHMETTERLING to a speed close to  $M = 1.0$ . Because of its aeroplane-like

construction and the lack of knowledge of the conditions at transonic speeds, the maximum speed was kept below  $M = 1.0$  by means of a MACH number regulator, which controlled the speed through the drive motor.

It is important to emphasize that Professor WAGNER, in parallel with the missile development, designed extremely valuable remote control simulators, which greatly assisted the training of the Armed Forces in the use of Hs 293 and SCHMETTERLING.

In those days, the factors which still influence the concepts of rational air defence had already led to the development of the air-to-air rocket. Compared with the air-to-surface weapon, the air-to-air rocket is much simpler, since the rocket can be fired along a simple pursuit course, using the aeroplane merely as a gun platform. To deal with this problem, Professor WAGNER built a further member of the Hs 293, viz. the Hs 298, which was smaller than the SCHMETTERLING.

On the other hand, Dr. MAX KRAMER developed, from the experience gained in the development of FRITZ X, the wire-guided, air-to-air rocket X-4. X-4 was almost ready for action at the end of the war. A "smaller brother" of the X-4 was the anti-tank rocket X-7, also wire-guided.

Finally I should like to draw your attention to a field of development on which we worked during the war without reaching a satisfactory solution. The problem to be solved, on which both Professor WAGNER and Dr. VOGT worked independently, was that of the use, against ship targets, of special warheads (doing the job of torpedoes) carried by guided missiles.

After the Hs 293 turned out to be a success, the development of a guided missile to bring the warhead close to the water surface presented no difficulties, but extensive tests had to be made about the water-entry and running characteristics of the warhead. In this way, Professors WAGNER and MADELUNG were able to construct a warhead which had no motor, but which followed a short, stable, rectilinear course, below the water surface, until its kinetic energy was exhausted.

This was achieved by fitting so-called separation edges or rings — on bombs these are also known as reflector plates — at the nose of the body. In this way a reproducible cavitation space was caused, the inside walls of which re-attached themselves to separation edges at the rear. The warhead body thus became drift stable. This short, stabilized submarine runner made it possible to build into the implement the effective proximity-fuse systems used in torpedoes during the war. It was not possible to have a large-scale production of this type of remote-controlled missiles ready for action from aeroplanes against ship targets by 1945.

*Remark:* The discussion contribution which Mr. A. R. Weyl has given after Dr. Benecke's lecture is published in the *Zeitschrift für Flugwissenschaften*, Vol. 5 (1957), May-issue, under the title "On the History of Guided-Weapon Development".

# GUIDANCE AND CONTROL OF THE HENSCHEL MISSILES

HERBERT A. WAGNER \*

## 1. INTRODUCTION

Since the end of World War II, many publications on the German missiles have appeared, based on the drawings and materials unearthed by the Allies, and on information given by expert German personnel; these publications appear to be very complete. The task of preparing a summarizing report on the co-ordination of guidance, control and airframe for the German missiles is, however, not an easy one. First, there is the great number of developments and methods. Then, since the German industry did not prepare technical progress reports, the post-war publications are more descriptive of the final article and do not dwell on development problems. In particular, the many missiles which did not reach tactical use can hardly be judged with justice by someone not directly connected with their development. Therefore, I will restrict myself to missiles which were used in service, and in particular to the HENSCHEL missiles.

## 2. OPERATIONAL MISSILES

If we disregard uncontrolled rockets, four missiles were used in operation. VON BRAUN'S V-2 is in a class by itself and cannot be compared with the other weapons. This supersonic rocket reached a remote target after flying through space. The gigantic task of development was undertaken with an equally gigantic effort, the approach being as modern as the task of the V-2. However, a report on the control and airframe problems of the V-2 should be presented by someone who was closer to this weapon.

The other three missiles which reached the front, namely the FRITZ X, the Hs 293 and the V-1, were all subsonic and used more conventional techniques.

The unguided V-1 (Fig. 1) was the last to be started. The experience gained by ASKANIA in automatic control pilots for unpowered aircraft, in which he had been interested for many years, was of great importance in the development of the V-1. The airframe had elevator and rudder, but no aileron. Very fast-responding pneumatic servos permitted gyro-stabilization of the airframe in both pitch and yaw. Considering the long solution time permitted for the V-1, such a tight stabilization was not really needed, except perhaps for launching. On the other hand, this artificial stabilization made the control problem quite

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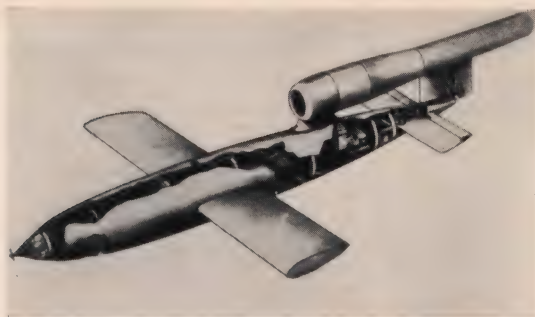


Fig. 1. V-1

insensitive to airframe properties — except for roll — and corresponds to the design of the control systems of many modern missiles. Although straight, automatic flight of unmanned airplanes had been made long ago, this development — mainly under the direction of Mr. LUSSEK — was perfected in an astoundingly short time.

The air-to-ground guided bomb FRITZ X was started by Dr. KRAMER in 1939 and its outstanding feature was its simplicity. The FRITZ X (Fig. 2) was a



Fig. 2. Fritz X

cruciform-wing missile, roll stabilized by means of one free gyro and ailerons located on the tail. Pitch and yaw control were achieved by spoiler-type elevators and rudders which moved directly in phase with radio-transmitted pulses. After being dropped from the bomber aircraft, a controller observed the bomb, which carried a flare. By means of a control stick and the radio link, he

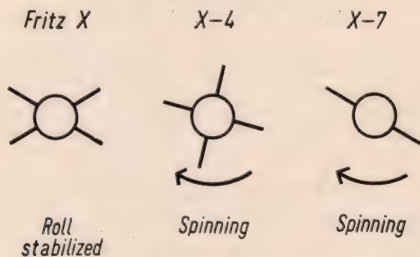


Fig. 3. Kramer's designs

could give up-and-down and left-to-right corrections, to maintain the bomb on the target sightline until it hit. The missile was successful, and is famous for having sunk the Italian battleship, *ROMA*.

In later designs, in order to reduce the airframe accuracy requirements, Dr. KRAMER dropped the roll stabilization (Fig. 3). He let the X-4 missile spin at a moderate rate and used the gyro only for phasing the signals properly. He allowed the ground-to-ground mono-wing anti-tank missile X-7, which needed only elevators, to spin. Dr. KRAMER certainly was the "master of simplicity", and I am sometimes curious to know the hitting accuracy achieved with the X-7.

### 3. THE BEGINNING AT HENSCHEL

I would now like to describe the beginning of the missile development at the HENSCHEL FLUGZEUGWERKE (HFW).

Very early ASKANIA had made suggestions for auto pilots for unmanned aircraft, and also SIEMENS had followed this line before the war. The DVL (DEUTSCHE VERSUCHSANSTALT FÜR LUFTFAHRT), a Government agency, had developed a model airframe for testing such auto pilots, with the intention of later developing a missile. It was intended to obtain control by elevator and rudder only. The SCHWARZ PROPELLER WERKE began to build these model airframes in 1937, but uncontrolled test launchings from an aircraft were unsuccessful and models with an auto pilot were never flown. At the end of 1939, the RLM (REICHSLUFTFAHRTMINISTERIUM) decided to place this development with the aircraft industry. HFW was interested and, upon the suggestion of Dr. LORENZ of the RLM, the HFW directors HORMEL and FRYDAG invited me to join HFW in January, 1940. We inspected the models built by SCHWARZ, and decided that the causes for the unsuccessful test launchings were the large dihedral, together with the side-slip angles at launching.

Before discussing the further developments at HENSCHEL, I would like to mention the key personnel, at HENSCHEL or outside, who participated in the early program and who are responsible for the early decisions.

The man in the RLM who supervised our developments, and who was later in charge of all Air Force missile development, was RUDOLF BRÉE. I am grateful for his competence and wisdom in guiding our developments with a minimum amount of "red tape" and paper work. In my department, JOSEF SCHWARZMANN was in charge of all electrical developments. He made the first design of the electric system for the Hs 293. REINHARD LAHDE and OTTO BOHLMANN were my best aeronautical advisors. WILFRIED HELL was one of the engineers we took over from the SCHWARZ PROPELLER WERKE. At the STASSFURTER RUNDFUNK GESELLSCHAFT, THEODOR STURM not only developed the radio receiver, but was always ready with excellent advice on our electrical problems.

We first started looking for an important, but soluble, problem. Without having any information on other missile developments, the following tasks were discussed.

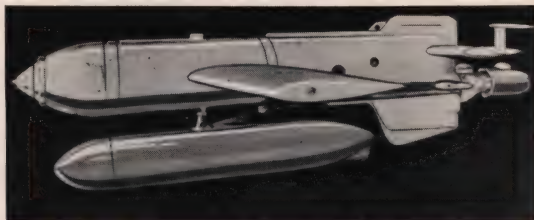
The RLM suggested a guided missile especially designed for attacking ships. This missile should glide toward the surface of the water, and then fly above it at low height automatically, by means of an altimeter. We disliked this task as being too difficult, particularly when rough seas were considered. We also

discarded, as being too difficult, the development of a torpedo which, after a guided flight, entered the water and was propelled under the surface. A further weapon discussed briefly at HENSCHEL, was an unguided missile such as the later V-1, to be used in long-range attacks on large target areas.

During my previous work at JUNKERS, the RLM had impressed upon me the need for thrift, the German resources being limited. At JUNKERS I also realized that a bomber was no better than the hitting accuracy of the bomb. We decided, therefore, to develop a glide bomb which could be guided by remote control with sufficient accuracy to hit directly a small target, for instance a ship, from a safe distance. Primitive simulator tests indicated that it was possible to solve this problem. The RLM accepted this proposal and named the design the Hs 293.

#### 4. DESCRIPTION OF THE Hs 293

This missile is shown in Fig. 4. The two wings were attached to the 550 kg warhead, and the rear part contained the instrumentation. The horizontal stabilizer was lifted above the wake of the wing. A set of five flares was located



*Fig. 4. Hs 293*

at the tail to make it visible to the controller. The addition of a WALTER boost rocket was the only major design change introduced as a result of the flight tests. Burning for 10 sec, and giving the missile an additional speed of 55 m/sec, it made low-altitude attacks possible. This became important when the Allies used radar to detect the attacking bombers.

#### 5. METHOD OF ATTACK

An attack is shown in Fig. 5. After flying toward the target, the bomber made a turn, and on resuming level flight, the Hs 293 was dropped. Less than a second later, the rocket was fired and the elevator pulled. When the missile came into the view of the controller, he moved a joystick to guide it, by radio commands, onto his target sightline, and maintained the missile in coincidence with the target until it hit. Depending upon the altitude of the attack, the final speed of the missile was between 120 to 250 m/sec. It flew for a period of time of 25 to 80 sec, which was sufficient for the controller to carry out his task.

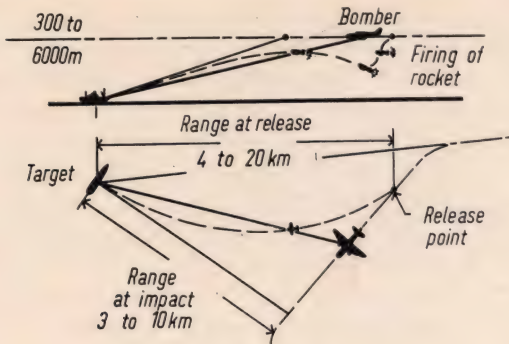


Fig. 5. Hs 293. Attack

## 6. SIMULATOR

Let us now discuss, in essentially historical sequence, the design problems of the guidance, of the automatic pilot, and of the airframe of the missile.

We started the design studies by building a simple guidance simulator consisting of a steel ball that could roll on a smooth glass plate (Fig. 6). A

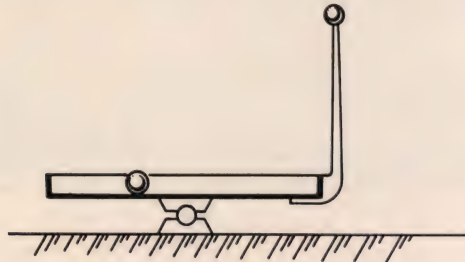


Fig. 6. Principle of guidance simulator

joystick was mechanically linked to this plate so that a man could incline it and thus control the acceleration of the ball. When a proper sensitivity was provided between the plate and the joystick, the man could, after moderate practice, keep the ball on a target area that corresponded to a medium-sized ship seen from a distance of 5 km; this distance was then still beyond anti-aircraft gun range.

## 7. GUIDANCE AND CONTROL SCHEME

Without waiting for the design and construction of the later model of the simulator, which permitted the introduction of details of the missile response, we chose the following scheme. The controller had a joystick (Fig. 7), the end of which could be moved in a vertical plane. The operator controlled the direction and magnitude of the lift of the mono-wing missile, by remote control of the bank angle and elevator deflection angle, respectively (Fig. 8). The plane in which the handle of the joystick could be moved corresponded approximately

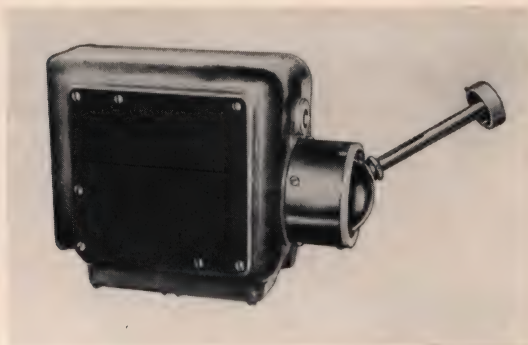


Fig. 7. Hs 293. Joystick

to the plane of vision. By moving the handle in this plane, the man would change the end point of the lift vector, without being aware of the mechanism resolving this motion into bank angle and lift magnitude, and without becoming aware of constant accelerations such as were caused by gravity and by the average curvature of the missile path.

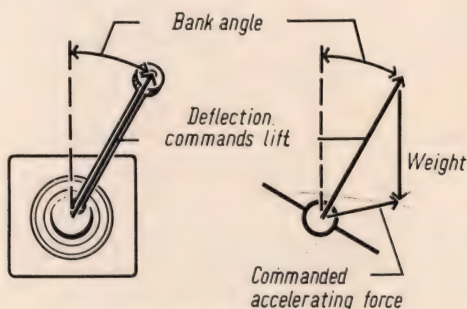


Fig. 8. Hs 293. Control principle

The RLM introduced us to a number of automatic pilots which could stabilize a missile, and make it follow a straight path. We were not interested in stabilization, but designed an auto pilot to provide a fast response of the bank angle, and the elevator deflection, to the respective radio commands.

## 8. STABILITY

Not having extensive simulators, we used the following reasoning with regard to the stability. In a small and fast airplane, the pitch oscillation — practically determined exclusively by the flow forces — is very fast as compared to the gravity-affected phugoid. The period of a pitch oscillation is less than 1 sec, that of the phugoid more than 100 sec. Our tests with the simple simulator had indicated that the frequency of human tracking — under the chosen conditions and methods — was well between these two frequencies. We concluded that it was unnecessary to bother about phugoid stability, any tendency toward instability being overridden by the remote guidance. Similarly,

lateral stability of the uncontrolled airframe was not considered to be a pertinent problem.

Since the pitch frequency is much faster than the attainable and required frequency of the guidance loop, it was, from the viewpoint of frequency response and stability, not necessary to introduce automatic pitch stabilization. The time lag in elevator servo and pitch response was later introduced — in an approximate way — into the simulator, with acceptable deterioration of the guidance accuracy.

## 9. ACCURACY OF THE AIRFRAME

From a modern viewpoint, the use of a gyro or accelerometer to achieve the proper magnitude of the lift might have appeared to be desirable. In this respect we needed high accuracy. Fig. 9 shows that, for a mono-wing missile, the error in the lift must be smaller than the average lift. Otherwise, the directional correlation between a change of control stick position, and a change of acceleration, is lost. Since we wanted to remain far from such a condition, even

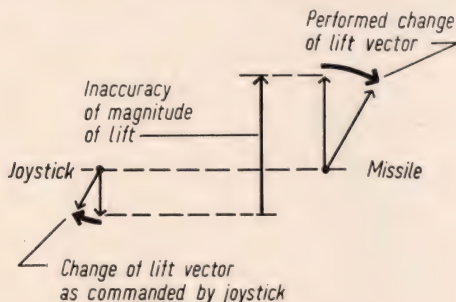


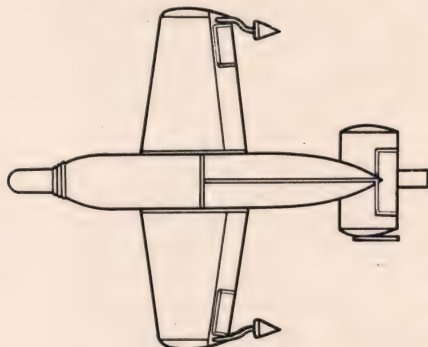
Fig. 9.

for the smallest average lift coefficient occurring in steep attacks, the inaccuracy of the elevator deflection had to be smaller than  $0.1^\circ$ . (The maximum deflection needed at slow speed was  $12^\circ$ .) The airframe was, from the very beginning, designed for inexpensive precision machining and assembly. All aerodynamic surfaces were symmetrical. Furthermore, we made a thorough study of the effects of surface inaccuracies on lift and also on rolling moments, and constructed a jig for measuring the contour of the surfaces by precision dial gauges. In production, the readings taken from these gauges were fed into a computer which told us how much to adjust the settings of the surfaces. This computer was suggested by Mr. ZUSE, who had been one of our employees. We gave him a contract to develop this computer. I believe this was one of the earliest applications of an electrical digital binary computer.

## 10. MACH NUMBER

In the design of the airframe, and the choice of the wing cross-section, we took great care to achieve a MACH-number-independent performance. An impact pressure plate influenced the elevator servo, so that the elevator

deflection was reduced in inverse proportion to the impact pressure. This alleviated the accuracy requirements for the radio link, and maintained a constant loop gain. Also, the zero-lift position of the elevator was automatically adjusted with MACH number. Behind each wing tip, a conical body was mounted

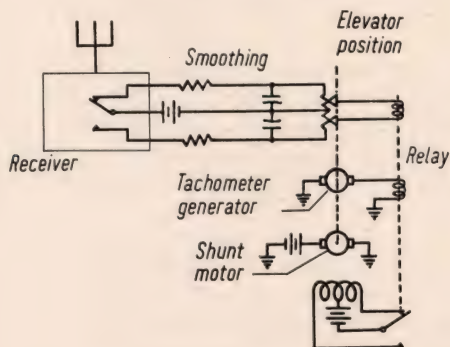


*Fig. 10. Hs 293 with drag bodies*

having a well designed, blunt front surface so that at MACH numbers of approximately 0.75 to 0.8, the drag began to rise rapidly (Fig. 10). Thus the airframe, even in steep attacks, would not exceed the highest MACH number for which it was designed, i. e. 0.85.

## 11. ELEVATOR SERVO

Since a detailed description of the elevator servo has been published, only a simplified schematic diagram is shown in Fig. 11. By means of a dithered SIEMENS T Relay, the time average of the field current of the elevator d. c. shunt motor was regulated in proportion to the difference of the elevator position, and the radio-transmitted electrical signal signifying the desired position. This difference was measured by means of potentiometers. A tachometer attached to the motor created a current used for damping the system.



*Fig. 11. Hs 293, Pitch control*

## 12. ROLL CONTROL

A rudder is not needed for the control of an aircraft in high-speed flight. We were certain that the omission of a rudder would be the key to a straight-forward success since moderate-sized ailerons can roll a small airframe in as short a time as the airframe can change its lift due to elevator deflection. Within an angular range of approximately  $\pm 20^\circ$ , the closed-loop roll frequency was approximately 2 c/sec. Larger changes of roll angle were performed somewhat, but not much, slower. The accuracy of the airframe was high enough to achieve equal rates of roll in both directions.

The directional co-ordination between a guidance error visually observed by the operator, and his manual reaction, is a psychological problem, particularly when the bomber makes a steep turn during missile guidance. After careful tests we decided on a fixed joystick box in the bomber, and on a free gyro with a vertical axis as the directional roll reference in the missile. To a guidance error which the observer sees as being upward-directed, he responds by moving the joystick in a direction he feels to be upward (Fig. 12).

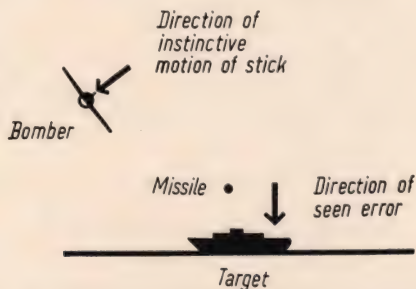


Fig. 12. Hs 293. Motion of joystick with banked aeroplane

The Hs 293 had conventional ailerons, bang-bang operated by magnets. The roll control system was similar to, but more complicated than, that of the SCHMETTERLING which I will show later.

## 13. FLIGHT TESTS

The flight tests started before wind tunnel test results were available. The first missile, launched on the 16th of December, 1940, was a failure because left and right were interchanged. The second missile, launched two days later, passed so close above the target — a small barn — that it was considered to be a complete success. Also, a considerable number of the missiles launched in the following months came close to the barn, or hit it. The 5,000 ton ship later used as a target was soon a sieve. Thus the flight tests proved the correctness of the design. However, numerous missiles failed for the following two reasons: First, the original white flares were not easy to see against a white background, and we had to change to red smokeless flares. Second, the electrical system often failed, partly because of minor design weaknesses, but particularly because of tube failures. Before going into production, all tubes were eliminated from the

missile auto pilot and were replaced by SIEMENS T Relays. I think this was done on the advice of Mr. STURM.

Early in the flight test period, Captain HOLLWECK was assigned by the Air Force to the Hs 293 project, and he, and the detachment under his command, were highly instrumental in gaining experience in flight testing, storage, maintenance and handling. Selected airmen were trained on the guidance simulator, and were then given the opportunity to make three Hs 293 launchings. The second, and particularly the third, launchings were, as a rule, successful.

#### 14. THE RADIO AND WIRE LINKS

I shall now make some remarks about the command transmission link. When we were starting the Hs 293 development early in 1940, the RLM informed us of the availability of a 6 m radio link, suitable for command transmission. But because this wave length was easy to jam, we requested that the RLM should order the development of a highly directional radio link using much shorter wavelengths. Simultaneously we started the development of a wire connection between aircraft and missile. After years of experimentation, and after decisive contributions by Dr. KRAMER of FRITZ X fame, this wire system worked reliably.

#### 15. TACTICAL USE

After the missile had been used successfully in the Gulf of Biscay and in the Dodecanese Islands, later attacks in the Mediterranean became less and less successful. Attacks from too long a range, and the unreliability of the missile were deemed to be responsible. A close inspection indicated that the wiring in the parent aircraft was flimsy and caused electric shorts. This deficiency was removed, but soon only night-time attacks were effective. Finally, the air superiority of the Allies became so great that bombers could not fly at all.

The Air Ministry had had two aircraft equipped with equipment for the detection of enemy jamming, but one of these aircraft was shot down early and jamming was never detected. Therefore, the wire link, which was at the front, ready to be used, was never employed.

When I came to the United States after the war, I was informed that jamming was used early after the first attacks. Therefore jamming and air superiority were the effective counter-measures against this missile, but the task chosen for it, and the methods of solution are sound and still up-to-date today.

#### 16. Hs 294

The Hs 293 as described was the only HENSCHEL missile which was used in operations. Of the other HENSCHEL developments, I will shortly describe only what I feel to be important.

Early in 1940, we had started tests on the shape of projectiles which could enter the water, without noticeable deflection, at shallow angles. The results of these studies were the torpedo bombs Hs 293 C and the larger Hs 294 (Fig. 13).

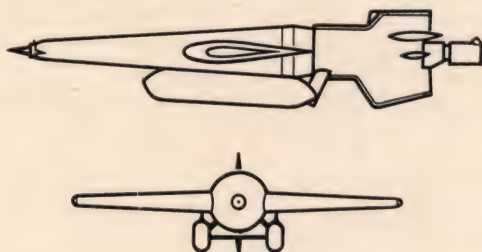


Fig. 13. Hs 294

They had slender conical warheads with an ogive nose. To bring the centre of gravity forward, the nose was made from solid steel. In order to stabilize these bodies under water within their cavity, a somewhat larger cone angle was used at the tail. We could curve the underwater path slightly upward by means of a small ridge on the upper side of the ogive. These missiles were supposed to be guided to a point in front of the water line of a ship. At water entry, the wings and the fuselage broke off at places where the supporting structure was weakened by sharp grooves.

## 17. BAR CONTROL

One of my assistants in the Aeronautical Institute of the TECHNISCHE HOCHSCHULE in Berlin, Dr. VON BORBELY, had made a theoretical study of the flow around airfoils. One of the results was that a very small control surface at the trailing edge of an airfoil, if deflected by  $90^\circ$ , would cause an astoundingly large increase of lift. Although this result was derived for frictionless flow, we tested it in the wind tunnel and were very surprised that the test results were comparable with the theory, even when the tests were repeated at large REYNOLDS numbers. Fig. 14 shows such a "bar" control surface. We then began to use such bars ("Leistenruder") on our missiles, viz., newer versions of the Hs 293, and the air-to-air missile Hs 298, and the ground-to-air

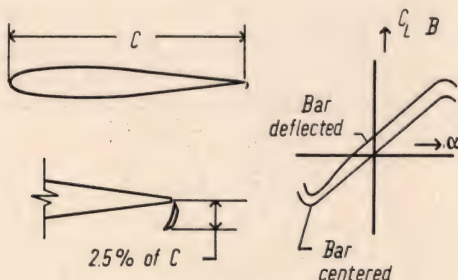


Fig. 14. Bar control

missile SCHMETTERLING. Tests performed in the high-speed wind tunnel of the DVL, indicated no noticeable decrease of control effect up to the highest MACH number achievable in this tunnel ( $\approx 0.88$ ).

## 18. "ZITTERROCHEN"

Dr. VOEPL made design studies for supersonic versions of our missiles, and carried out tests in the wind tunnel in Göttingen. A result of these tests was the model known as ZITTERROCHEN, which had a round fuselage and triangular wings. It was found that the wing plan-form shown in Fig. 15 gave the same

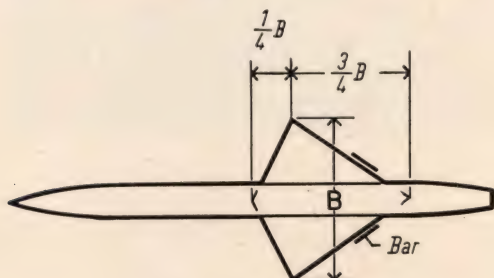


Fig. 15. "Zitterrochen"

position of the neutral point at high subsonic, and at low supersonic, MACH numbers. The sonic range could not be covered in available wind tunnels. The trailing edge of this wing made an acute angle with the direction of flight. Bars at the root end of these trailing edges gave effective lift control for this tailless, mono-wing design, within the entire tested range of MACH numbers from subsonic to approximately  $M = 1.5$ . We intended to use the ZITTERROCHEN design for our supersonic missile projects.

## 19. Hs 117

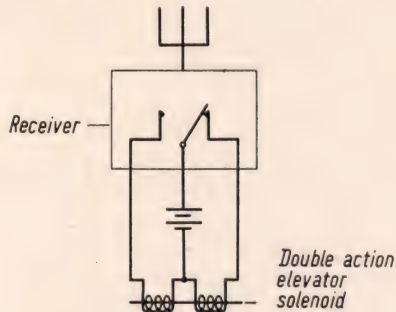
I would like now to describe the main features of the Hs 117. We had unsuccessfully proposed this ground-to-air missile in 1941, but in 1943 it



Fig. 16. Hs 117

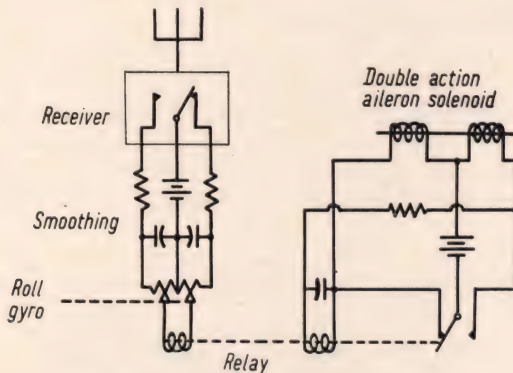
received highest priority and it was, with the exception of the X-4, the only remotely guided missile with sizeable production orders at the end of the war. The project engineer of the Hs 117 was Mr. HENRICI.

In order to alleviate the problems at high subsonic speed, we used swept wings (Fig. 16). However, we had to produce a new design, based on extensive wind tunnel tests, before the airframe became as MACH number insensitive as that of the Hs 293. We regulated the thrust of the liquid-propellant sustainer rocket, so that the speed of the missile was maintained at a constant MACH number of 0.84.



*Fig. 17. Hs 117. Pitch control*

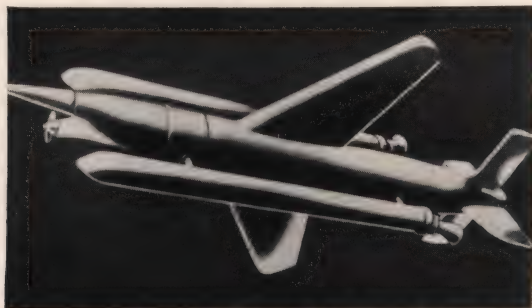
Visual sightline guidance by means of radio commands was used as for the Hs 293. A wire command link was not feasible. The use of bars as control surfaces, and the skill and experience gained in the past, led to a simpler and miniaturized missile auto pilot. Fig. 17 shows that the elevators were moved directly by the radio-transmitted signals, as in the FRITZ X. The roll control system is shown in some detail in Fig. 18. The pulses indicating, by their average



*Fig. 18. Hs 117. Roll control*

duration, left and right commands were transmitted by two corresponding audio channels of the radio link. The time-average voltage was formed by RC filters. These voltages were compared with the roll angle of the missile by two potentiometers mounted on the free roll gyro. Any deviation from the desired roll attitude, caused a SIEMENS relay to close a contact which energized

the magnet operating the proper aileron bar. Damping of the roll attitude, and dithering of the relay, were achieved by feeding the magnet-operating pulses — delayed by an RC circuit — to another coil of the same SIEMENS relay. This simple damping circuit was suggested by an engineer of Peenemünde, whose name I have forgotten.



*Fig. 19. Hs 117 with boost rockets*

We used zero-length launching. One solid-propellant boost rocket was mounted below the fuselage, the other one above (Fig. 19). The lower rocket was ignited first. In order to make the flight path of the missile insensitive to the inaccuracy of time interval between the ignition of the two rockets, and to differences of burning rate, the thrust of the rockets was directed with great precision through a point somewhat behind of the centre of gravity of the missile. The boost rockets were jettisoned after 4 sec, and the sustaining rocket was ignited. Based on the test launchings with the Hs 117, and on experience with the Hs 293, all concerned felt certain that this weapon would be successful, and large-scale production was initiated and in progress, in spite of the fact that the motor of the sustainer rocket was not yet operating at the desired degree of efficiency. Furthermore, there was not the slightest doubt of a sufficient number of enemy targets. However, in spite of the use of 40 cm wavelength for the radio link, there is scarcely any doubt that jamming would soon have made this weapon unsuccessful.

Of the various homing heads under development, the infrared homing head developed by Dr. KUTZSCHER was the most suitable one, but it would not have come early enough to change the tide.

## 20. REMARKS

When we started missile development at the beginning of the war, there were no missile weapon system engineers in Germany. The early HENSCHEL designs were produced by careful discussion of the various problems and solutions by a small group of experienced engineers. When, after the first success, contracts poured into our development department, designers of less experience were heading some projects and the discussions were no longer so broad. In missile design, even a small mistake may have severe consequences. I would like to mention one example. For the underwater fuse of the Hs 294, we issued a written specification that this fuse would have to stand 400 atm of water

pressure. The personnel working on this missile did not appreciate the importance of the specification and missed testing the fuse accordingly. When flight tests with the missile indicated that the fuse did not function properly, extensive test facilities had to be created, and one and one-half years were lost in locating and remedying the trouble. The Hs 294 was not used in combat.

Another interesting sidelight: When we consider that the most complex part of the V-1, namely the auto pilot and the servomechanisms, was based on designs which ASKANIA had started long before the war, we can say that all the developers of successful missiles had started before, or at the beginning of the war. These same agencies were also the only ones who had sizeable production contracts at the end of the war. At the beginning, they could choose the most important and/or the simplest tasks. In solving these problems, they gained experience in co-ordinating the elements of the missile weapon systems, i.e. purpose, sensing device, auto pilot and airframe, so that their later designs had sufficient superiority to survive.

## 21. CONCLUSION

Although missiles were one of the spectacular technical developments in World War II, they were not decisive. When they became operationally available in the second half of the war, they were not powerful enough to alter the situation. As far as guided missiles are concerned, the most important lesson is that one should not under-estimate an enemy's skill in interfering with radio techniques.

## DISCUSSION

Question (anonymous): Are there any data on the length of the stable under water path of the Hs 294 as related to the speed at entry?

Prof. Dr. WAGNER: After some tests at the PREUSSISCHE VERSUCHSANSTALT FÜR WASSERBAU UND SCHIFFSBAU, we constructed our own model tank and a gun — operated by compressed air — to shoot the models. The drag coefficient was approximately .06 with reference to the largest frontal area (largest diameter) of the Hs 294 bomb. This gave an under water path of the Hs 294 of 60 to 80 metres. This should correspond to velocities at water entry of approximately 150 to 180 m/sec. The small ridge on the upper side of the ogive — as mentioned in the lecture — helped to increase the usable length of the under water path.

Dipl.-Ing. HESKY (Dortmund): Have the changes introduced into the aerodynamic design of the Hs 298 been successful?

Prof. Dr. WAGNER: I am sorry that I cannot remember the final results. On the one hand I believe to remember that the changes were successful. On the other hand, I believe to remember that at least guided flights were not made with the streamlined model of the Hs 298.

Prof. Dr. QUICK (Aachen): You had mounted several bodies on the Hs 293, which had the purpose of limiting the MACH number. I remember distinctly that these drag bodies were tested in a wind tunnel, and I would like to know whether or not these bodies really did function?

Prof. Dr. WAGNER: The wind tunnel tests with these drag bodies were very successful. We flight tested a number of Hs 293s with drag bodies attached. However, the problem never became urgent enough for a careful investigation. The weather conditions at Peenemünde which forced the FRITZ X to perform the tests in Italy did not permit high and steep launching of the Hs 293. The development of the radar defense of the target ships forced us to make low attacks in combat. Therefore the drag bodies were never used in combat.

# CONTRIBUTIONS TO THE GUIDANCE OF MISSILES

EDUARD FISCHEL \*

## 1. INTRODUCTION AND PROBLEMS INVOLVED

During World War II, the DEUTSCHE FORSCHUNGSANSTALT FÜR SEGELFLUG (DFS) worked intensively on the problems relating to the remote control of missiles — a fact not unknown to the enemy. In June 1945, during the first tour of inspection of undestroyed equipment undertaken by an Anglo-American commission, I was informed that no bombs had been dropped on Ainring, in order that the equipment could be obtained intact for study purposes.

At that time the Allies were preparing a full-scale assault on the Japanese mainland, and it was important for them to know if the Japanese had had the opportunity of seeing the latest developments. They were informed that probably no Japanese commission had seen Ainring, and definitely had not inspected the work on guided missiles carried out at my Institute.

I have tried to compress, within the narrow limits of this paper, the work done at the Institute on the remote control of missiles. I can touch only on the main points, leaving details aside, since I no longer have all the data and must draw on my memory for many of the facts. I trust, however, that the picture I have drawn will not appear too kaleidoscopic.

In considering the problems involved we realized that the control procedure was a regulatory process in which man played a part, and that the controlling action depends on the psychological make up of the bomb-aimer. Therefore, for the purpose of investigating suitable control methods and establishing a criterion for the selection of suitable bomb-aimers, we decided to build models which would take this into account. The bomb-aimers were to be trained on the model. I was well aware of the value of such training methods, since in New York during 1938 I saw the Link Trainer in which American pilots were trained and kept in training. There were two ways in which we could approach the design of the new equipment:

1. A true, miniaturized representation of the flight path with the aid of suitable equipment or
2. A mechanical structure, of any form whatever, which would continuously give the bomb-aimer the same impression that he would have if he were actually performing the control operations.

\* Prof. Dr.-Ing. — Formerly: Chief of the Institut für Flugausrüstung der Deutschen Forschungsanstalt für Segelflug (DFS), Darmstadt, transferred to Ainring, Upper Bavaria, during World War II. — At present: Chief of the Navigational Instrument Laboratory of Kearfott Company, Inc., Klifton, N. J.

The year before, Professor WAGNER (HENSCHEL AG) had shown us what was probably the simplest design in the second group. In this arrangement the bomb-aimer used a linkage arrangement to control a plate in such a way that a ball placed upon it remained in the middle, or could be brought to any other point on the plate (see Professor WAGNER's paper in this book).

We decided, however, to build devices in accordance with group 1, since it was our intention to carry our investigations through to the smallest detail. We did not then suspect just how all-encompassing the field would be, nor that a computer which could simulate all the mechanics of flight would result.

In most cases it was possible to solve the individual mathematical problems as the model tests proceeded.

We divided the control methods into three groups:

1. Line-of-sight method,
2. Control by means of a TV image,
3. Beam-riding method.

## 2. CONTROL OF THE GLIDER BOMBS

### 2.1. Line-of-Sight Method

This method derives its name from the fact that the bomb-aimer, in order to hit the target, keeps it in coincidence with the bomb while the latter is on its way to the target\*.

We shall now discuss what the bomb-aimer does in order to maintain this coincidence, or how he brings the bomb back into coincidence. He operates the rudders through a radio set, by means of a control stick, and the bomb begins to describe a curve similar to the arc of a circle (Fig. 1). The bomb-aimer sees the bomb move faster and faster in the commanded direction, and he recognizes it as an accelerated motion, but before it reaches the line of coincidence he must decelerate it by opposite rudder deflection. He recognizes this as a retarded motion and he must initiate the counter-action sufficiently soon to prevent over-control. Since the bomb appears to the bomb-aimer to be always accelerating, this type of control is also referred to as the "acceleration method".

By automatic transformation of the command, this method can be changed to one which straightens out the initial arc and therefore gives the bomb-aimer the impression that the bomb is approaching the target at a constant velocity. If the rectified part of the control process is large as compared with the two end arcs, the method may be referred to as a "velocity method". In this case also the bomb-aimer must decelerate the bomb at the proper time.

Studies and tests on models had shown that the acceleration method made maximum use of the bomb's control capabilities and allowed the largest flight path curvature. It required considerable training, however. In more or less straight bomb runs, the velocity method could be used, and it required less training. To meet the multiple requirements of practice, H. WAGNER based the control of the HENSCHEL bomb essentially on the acceleration method. It was so

\* Frequently also termed the three-point method, because the bomb-aimer, the bomb, and the target all lie along the same straight line.

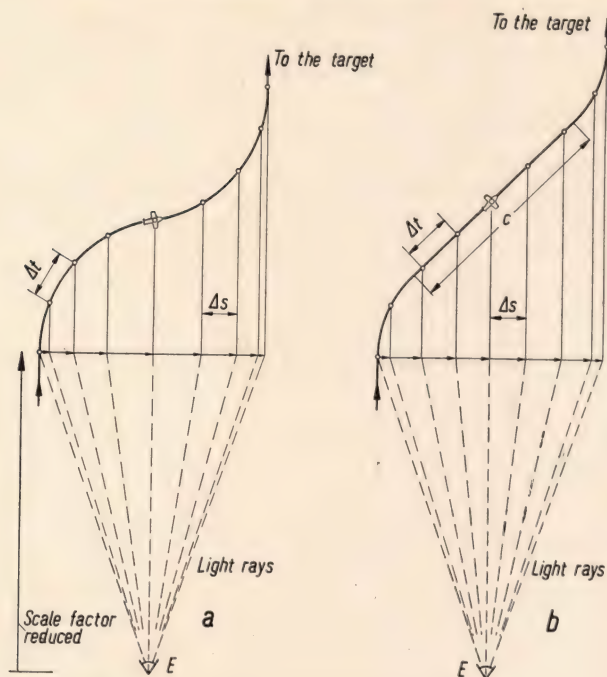


Fig. 1. Control methods

a) Pure acceleration method,

b) Mixed velocity method,

$E = \text{Eye of the bomb-aimer, } \Delta t = \text{const, } \Delta^2 s / \Delta t^2 \approx \text{const.}$

In section c pure velocity method ( $\Delta^2 s / \Delta t^2 = 0$ ).

designed, however, that a velocity portion could be added to it if allowed by the approach path.

## 2.2. Glider Bomb Model Installation

In order to study the control method, we built a model of the flight path to a scale of 1 : 1000. The time scale was not shortened (Fig. 2).

The bomb itself was represented by a small incandescent lamp, having a diameter of approximately 2 mm, carried by a car, the so-called "bomb car". This was driven by a motor and it could be steered along the floor of the testing room, thus giving a ground velocity to the bomb. It was provided with a mast strung with a thin wire, to which the bomb was attached and moved down at a controlled velocity. The course of the car, and the vertical velocity of the bomb, could be set and varied from the bomb-aimer's position. The path of the bomb was thus determined by its two velocity components and the course of the bomb car. The lamp was heated through the suspension wire, and could be clearly seen by the bomb-aimer from a considerable distance.

The bomb-aimer sat on a second car (the aeroplane car), also driven by an electric motor, which could be controlled with one hand. In front of him was a control stick, which sent signals over a cable to the bomb car and controlled

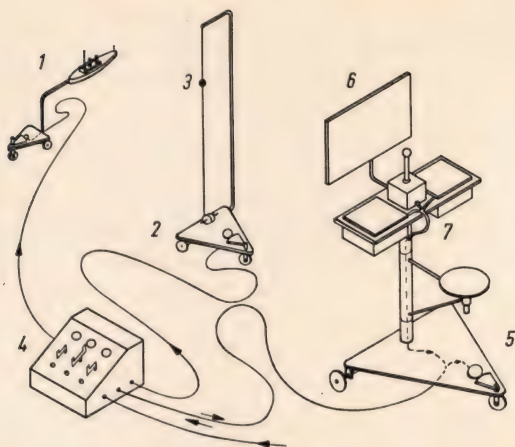


Fig. 2. Model system for glider bombs

- 1 = Target car,      2 = Bomb car,  
3 = Model bomb,    4 = Control panel,  
5 = Aeroplane car,   6 = Screen,  
7 = Interchangeable parts for course and altitude.

the course and the downward velocity of the bomb. A screen was placed in front of the stick in order to prevent the bomb-aimer from seeing the body of the bomb car, for otherwise he could easily judge the distance to the target by reference to the ground. He would thus avail himself of an expedient which would not be at hand in practice.

The target, in the form of a small model ship, was also carried on a car, the "target car". The course and speed of the target car could be controlled from an independent station, the control panel.

The control panel was used to switch the whole system on or off, and also to record the movements of the stick and bomb.

In order to investigate the various methods (acceleration, velocity, and a combination of both) interchangeable units were mounted to the right and left of the stick. These units provided the proper characteristics for the respective control operation. The flight characteristics of the bomb were given no particular attention, except that course changes were not made in a jerky, but in a smooth, exponential manner.

The device was demonstrated repeatedly to the Air Ministry, the test stations of the Air Force and to industry, and it was used to train bomb-aimers. The testing stations later received their own equipment.

### 3. THE CONTROL OF BOMBS BY TELEVISION

In order to improve accuracy we investigated two methods which basically are very similar. In one method a television transmitter is mounted in a bomb, and the bomb-aimer directs the bomb in accordance with the transmitted picture. In the other method, a target-seeking device is used which directs the

bomb to the target automatically. In the latter case the bomb-aimer is eliminated and, according to our concepts, we have a control and not a guidance system. Both methods demand the same degree of manoeuvrability of the bomb.

Insofar as directing the television bomb is concerned, this is considerably simpler than in the case of the line-of-sight method. All the bomb-aimer has to do is to keep the target in the centre of his picture.

In the case of the coincidence, or line-of-sight method, we referred to an acceleration method and a velocity method. We might now refer to the television method as a "stationary method" because the bomb-aimer achieves coincidence by pure rotation and is not required to establish it by lateral movements of the bomb.

### 3.1. Theoretical Considerations

New problems arose which were investigated both theoretically and with models. If the target is in motion, or if there is a wind, the television bomb moves towards the target along a so-called "dog-leg curve". The curvature of this increases steadily in the vicinity of the target, and occasionally becomes so large that the bomb cannot follow. The bomb then flies off at its smallest radius of curvature and misses the target.

Some of the results of a theoretical study are shown in Fig. 3. The curves in the upper left-hand corner represent the approach paths of a television bomb directed against a ship that is moving towards the right. The attack is shown

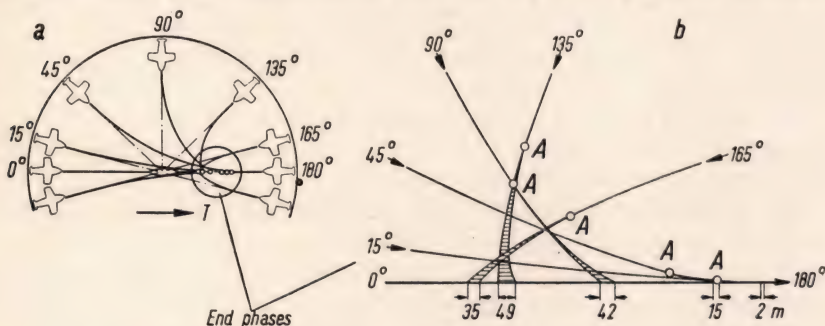


Fig. 3. Trajectories and their deviation from the "dog-leg" curves

a) The approach angles,

$\omega_{\max} = 5^\circ/\text{sec}$  for the bomb;

b) End phases enlarged.

$T = \text{Direction of motion of the target.}$

$V_{\text{bomb}}/V_{\text{ship}} = 5;$

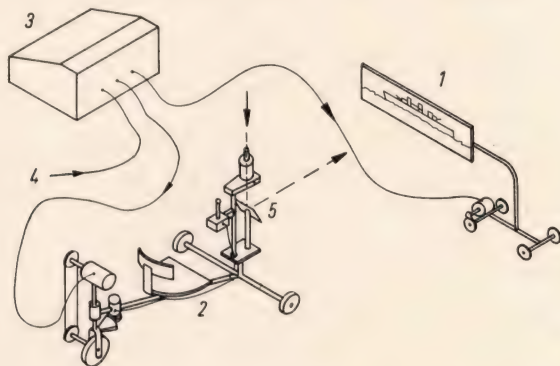
from a large number of different directions. The curves in the lower right-hand corner show an enlarged view of the end phases. At points A the bomb can no longer follow the increasing curvature of the curve and lags behind the target motion. Under the assumed conditions, an approach angle of about  $135^\circ$  is the most unfavourable and causes a deviation of 49 m.

It is, however, possible to minimize this error or even to eliminate it altogether. A simple method consists of training the bomb-aimer to aim the bomb in advance to the target, just as a marks-man aims his gun when shooting at clay pigeons. The accurate way is to provide him with some trick for finding

the collision course and holding to it. He then steers along a straight course towards a point at which both bodies will meet. If the ship under attack changes its course or speed, then a new course will have to be flown.

### 3.2. A System for Studying the Television Method

In addition to the theoretical investigations, we built and used a model for studying the bomb runs. A schematic representation is shown in Fig. 4. However, before I describe this model, I should like to invite your attention to one point.



*Fig. 4. Device for investigating the TV-method*

1 = Target car,    2 = Bomb car,  
3 = Control panel, 4 = Power supply,  
5 = Reflector parts interchangeable for

a) Path tangent method,    b) Air frame fixed method,    c) Collision course.

In the accelerated conditions of curved flight, or if the bomb makes a turn, its longitudinal axis no longer points along the tangent to the path. If now the bomb-aimer steers according to the television image, the instantaneous flight direction is displaced an amount equal to the angle of attack, the flight path degenerates and the miss-distance increases. This can be remedied by stabilizing the camera in the wind direction (tangent to path) or — and this is simpler — by coupling the camera's objective to a weather cock.

The model device had to be scaled down, it had to move along the same path as would an actual bomb, and the impression received by the bomb-aimer had to be the same as would be obtained in practice. The arrangement was simple. A direct picture and direct control were used instead of a television picture and remote control. For this purpose the bomb-aimer sat on a car (bomb car) which simulated the bomb and looked (Fig. 4), through a telescope, into a mirror from which was reflected an image of the target. Since the television image rotated with the actual bomb, a unit had to be incorporated in the optical system of the model which would achieve the same effect. To this end a DOVE-prism was used, which rotated whenever the bomb car described a curve. The details are not shown in the figure, but they consisted of a transmitting system from the control wheel to the telescope. The outline of a ship was selected as the

target, and this was also mounted on a car, the "target car"\*. The scale of the model was 1:400; the time scale remained unchanged.

The reflector parts were interchangeable so that the following various methods could be investigated:

1. The arrangement with fixed reflector corresponding to the case in which the camera is stabilized by the wind indicator, since the axis of the image and the tangent to the path always coincide.

2. The arrangement in which reverse rotation was provided corresponded to the case in which the camera is fixed in position. The reversing action was obtained from the rotational motion of the car, and represented the angle of attack.

3. The arrangement with gyro-stabilized reflector for flying a collision course.

Tests were performed, and the trajectories and errors determined with each of the above arrangements.

Method 3 yielded practically no errors. It was, however, more difficult to manipulate than method 1\*\*. Method 2 naturally produced the largest error. A circular sight reflected in the television image proved helpful\*\*\* in method 1†.

### 3.3. The Table Training Device

If we had intended to train bomb-aimers in the TV method, the car model just described would have produced excellent results. However, it required a large surface because of the low scale factor of 1:400, and was not as convenient as a suggestion for a table training device, shown in Fig. 5. The basic idea behind this was to project films on the screen, these films having been taken during attacks on a ship. The aircraft taking the films flew past the ship so that the target moved away from the centre of the screen. If such films are used for training purposes, the bomb-aimer must be provided with means for bringing the target back to the centre of the screen, and this actually represented the flight direction of the TV bomb. The figure shows two rotatable mirrors and a Dove-prism inserted in the path of the light rays.

The suggested arrangement was developed for a monoplane missile which was controlled only with elevators and ailerons. The controlled prism provides the bank positions, and the two mirrors the course rotation and the elevation. The control signals were generated in a computer which solved the flight equations of the bomb, and which was connected to the control stick.

Although the device was not built, it should be mentioned here because it represents a design which we frequently considered for purely training purposes, i.e. for glider bombs and anti-aircraft rockets.

\* In a second development, the ship's outline could be moved up and down, so as to force the bomb-aimer to guide the bomb in two planes. In this case the reversible mirror was mounted so as to be rotatable about the horizontal axis.

\*\* If No. 1 is to be defined as a stationary method, then No. 3 is a velocity method, because the bomb-aimer must keep the velocity components of the missile parallel to the motion of the target by guiding the course of the missile.

\*\*\* The objective used in the model was placed at our disposal by Henschel AG (Prof. H. Wagner).

† Herr Dipl.-Ing. Münster had quoted in his lecture a fourth system (inverse rotation system), which we also investigated. I had not mentioned it in my lecture due to lack of time, particularly since it showed no new criterions. But it is of considerable importance for the construction principles of the devices.

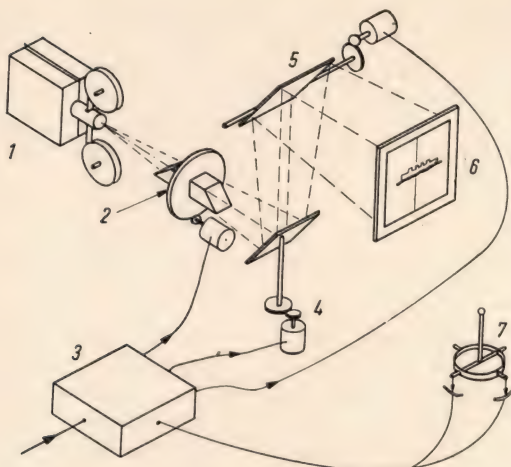


Fig. 5. Table training device for the TV-method

- |                     |  |
|---------------------|--|
| 1 = Film projector, | 2 = Rotation of the image about the longitudinal axis, |
| 3 = Computer,       | 4 = Course,  |
| 5 = Elevation,      | 6 = Screen,  |
| 7 = Control stick.  |  |

### 3.4. Comments on the TV Control Method

The main advantage of this method is that the accuracy increases as the distance to the target decreases. A decided disadvantage, however, is that it is useful only in the last stages since the resolution of the picture is not adequate at greater distances. It must therefore be combined with another system.

## 4. MODEL FOR THE BEAM-RIDING METHOD

We also built a model for this method, but I shall not describe it until I have made some remarks about the technique involved. It is substantially different from the methods described thus far. The bomb-aimer does not control the bomb directly; he merely directs the guide beam at the target. The bomb steers itself along the axis of the guide beam, and tries to maintain itself there if the beam wanders. We thus have a combined system consisting of a control method for the bomb, and a steering of the beam by the bomb-aimer.

It was felt at the outset that the directing of the beam would present no problem, since it involved only relatively slow rotation of the beam emitter with respect to the ground, or to the aircraft, and no masses had to be given translatable motion\*.

With regard to the problem of controlling the bomb in the beam, we had arrived at only an approximate solution, since an exact simulator required computers similar to those we built for the anti-aircraft rockets, and which we shall consider later on.

\* According to our previous terminology we would assign the name "stationary method" to it.

The main purpose of the device was to investigate a photo-electric cell control by means of a working model. If I recall correctly, the photo-electric cell equipment was furnished by the Marburg University. The set-up consisted of a swivel-mounted searchlight having an ultraviolet light source, and a "bomb car" provided with a drive and two photo-electric cells, which steered the car in the searchlight beam. The stability conditions could also be tested with this arrangement.

## 5. MODELS FOR ANTI-AIRCRAFT ROCKETS

The devices described so far were intended for glider bombs launched from aircraft against ground targets. Around the middle of 1943, we started the development of models of rockets fired from the ground against air targets. The order came from the Army Arsenal at Karlshagen (General DORNBERGER and Professor VON BRAUN), and the device was intended for the "WASSERFALL" rocket. We were instructed to investigate how effective the control method was, and then the bomb-aimers were to be trained. It was built along the same lines as the glider-bomb system shown in Fig. 2. As we shall see later on, the principle of the cars for the anti-aircraft rockets and the target were modified so as to meet the new requirements. The whole installation was made ready in the early summer of 1944, and was set up in a large building. Actual tests were begun, and besides the customer, numerous personnel from the Air Ministry and industry, who were engaged in anti-aircraft rocket development, observed the tests. I shall not describe the set-up, however, because it was merely the forerunner of an enlarged installation which had provision for taking into account all the flight mechanics of the rocket.

For the Karlshagen Arsenal, the controllability of the "WASSERFALL" rocket had to be more thoroughly investigated by introducing computers for the moment and force equations of the missile. This was not done in the first system. One of the main requirements was the representation of the instantaneous angle of attack, from which information could be obtained about the mechanical stresses in the structure. In addition, the model was to be used to study the initial phases through which the rocket has to pass after the start in order to establish initial coincidence with the target.

If we had set up the device along these lines we would have completely satisfied the customer's requirements. However, by retaining the built-up type construction of the previous devices, we had the opportunity of meeting the requirements of the other anti-aircraft rockets. Thus, from the very inception of the design, consideration was given to "RHEINTOCHTER" (RHEINMETALL-BORSIG) and to "ENZIAN" (MESSERSCHMITT). The main difference was that the "WASSERFALL" and "RHEINTOCHTER" were cruciform-wing missiles, and the "ENZIAN" a monoplane missile.

The new unit was almost completed by the end of the war. It was built to a scale of 1:5000 and had a combat space 40 km long, 25 km high, and 25 km wide. Fig. 6 shows the general construction of the computers, as set up for cruciform-wing missiles. The computers can be seen at the lower right, the model space at the upper left, and the aircraft control at the lower left.

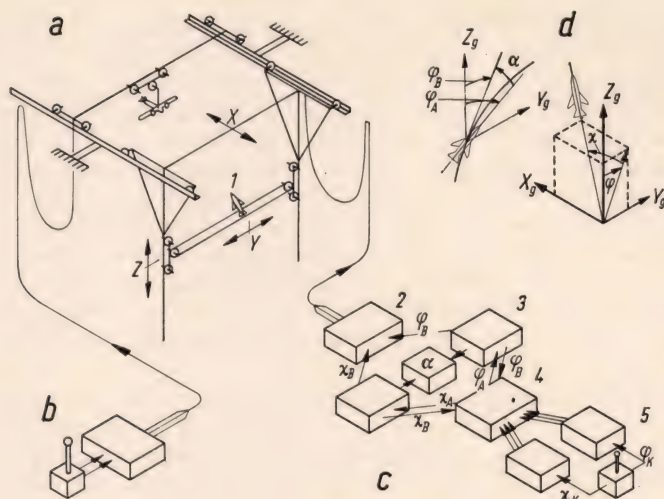


Fig. 6. Device for investigating anti-aircraft rocket design

- |                              |                            |
|------------------------------|----------------------------|
| a) Model space (Integrator), | b) Aircraft control,       |
| c) Rocket control,           | d) The angles used.        |
| 1 = Model rocket,            | 2 = Co-ordinate converter, |
| 3 = Path computer,           | 4 = Phantom set,           |
| 5 = Converters.              |                            |

The displacements  $\varphi_K$  and  $\chi_K$  of the rocket control stick were picked off as electric signals, and both differentiated and integrated in the "converters" \*.

The  $2 \times 3$  values obtained in this manner passed to the "phantom-set", which solved the rocket's moment equations in both planes. The inertia moment, the damping moment, aerodynamic restoring moment, and rudder moment were held in equilibrium. The rudder moment was a function of the displacement and rate of the stick. The integral turned the gyro pick-up. The output of the phantom unit were the angles  $\varphi_A$  and  $\chi_A$  of the rocket in space. For the sake of completeness, the system was originally designed to incorporate a model rocket, hence its name.

In the next two units, the "path computers", the two path tangent angles  $\varphi_B$  and  $\chi_B$  were determined from the force equations. The centrifugal force, the lift, and the components of thrust and earth's attraction were brought into equilibrium. Since the moment equations required the partial angles of attack, the angles  $\varphi_B$  and  $\chi_B$  were fed back continuously to the phantom device. The angle of attack computer received its information from the path computers, and from it formed the total angle of attack  $\alpha$ . A complete treatment of the problem would have required the determination of all forces in the flight direction. The customer had, however, abandoned the idea of instrumenting this, and had informed us how the respective values varied with time (fuel consumption) and altitude (ram pressure).

\* I may recall that the signal, originated by a movement of the stick, corresponds to an acceleration procedure, whereas the differentiation comes near to the speed procedure. The more one approaches the proper speed procedure, the more the steering range will be limited. To enlarge this range, one has to form the integral of the stick movement and to superpose it.

The next unit was the "co-ordinate converter", in which two operations were performed simultaneously. It received the angles  $\varphi_B$  and  $\chi_B$  of the tangents to the path in polar co-ordinates, and converted them into rectangular co-ordinates. It then multiplied them by the velocity of the rocket, which was given as a function of the flying time and altitude.

Finally, the three velocity components were integrated in the "integrator" to give distance. The device itself represented the combat zone of the model. Two parallel rails on which the rocket and aircraft cars rolled were mounted on the ceiling over this area. The rocket car, which moved in the  $X$ -direction, consisted of a U-shaped frame open at the bottom. Two small carriages moved up or down together at the same height in the  $Z$ -direction. They carried a winch arrangement that subsequently received the model rocket, and made motion in the  $Y$ -direction possible.

The aeroplane used as a target was also carried by a car which, however, was of simpler design. It moved in the  $X$ -direction on the parallel rails which supported the rocket car. It was a little higher than the latter, however, so that the two cars would not interfere with each other. A cable moved the target in the  $Y$ -direction and a winch in the  $Z$ -direction. The pilot had a stick, by means of which he steered the model. The signals were fed to a computer which was essentially a co-ordinate resolver, and which also imparted a certain amount of time lag to the commands, in order to take into account the limited manoeuvrability of large aircraft.

Another device that was developed was a "run-in computer" which derived from the co-ordinates of the rocket and the aircraft, and their velocities, a quantity which, when zeroed, directed the bomb smoothly into coincidence or line of sight. The method was primarily intended for use in one co-ordinate.

In concluding this enumeration of the systems developed, we might mention a recorder for plotting all the important quantities, i.e. stick motion, angle of attack, and flight path. The design of this device was assigned to the OTT COMPANY.

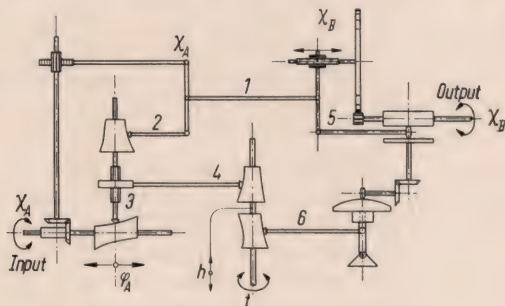


Fig. 7. Computer for the force equation in  $X$ :

$$\chi_A - \frac{1}{n'} \sin \chi_A \cos \varphi_A = \frac{v}{n'g} \chi_B' + \chi_B.$$

$$1 = \chi_A + \frac{1}{n'} \sin \chi_A \cos \varphi_A, \quad 2 = -\frac{1}{n'} \sin \chi_A \cos \varphi_A,$$

$$3 = \sin \chi_A \cos \varphi_A, \quad 4 = \frac{1}{n'}, \quad 5 = \frac{v}{n'g} \chi_B', \quad 6 = \frac{n'g}{v}.$$

To give you an idea of the mechanics of the system, I shall explain the principle of the computer for the force equation in the  $\chi$ -plane (Fig. 7). The force equation, arranged in a form to be used in the computer, is shown at the top. All the individual magnitudes are known except  $n'$  which represents a specific lift coefficient. The mechanisms are three-dimensional cams, ball and disc integrators, differentials, and the necessary links. The input quantities are the angles  $\varphi_A$  and  $\chi_A$  describing the position of the rocket, the altitude  $h$ , and the time  $t$ . The output is the angle of the tangent to the path  $\chi_B$ . The left-hand side forms the product, and carries out the addition, whilst the right-hand side integrates  $d\chi_B/dt$  to the angle  $\chi_B$ . The details will be clear to the expert from examination of the figure.

The US Air Force shipped the incomplete device to the States during the autumn of 1945.

## 6. COURSE AND RANGE CORRECTOR FOR THE V-1

In the subsequent development of the V-1, it was planned to improve its speed and range by using a more powerful power-plant (turbojet) and to increase its accuracy by means of a remote course and range corrector. In the autumn of 1944, my Institute received orders to undertake the remote-control part of the program. The Air Ministry provided the high-frequency equipment and the personnel. A method of navigation was proposed which I shall describe with the aid of Fig. 8. After being launched the V-1 turns, as previously, into the course directed by the compass, and covers the greater part of the course (heavy line). When it reaches  $A$ , a built-in transmitter emits a pulse, which is received by two stations  $B_1$  and  $B_2$ , and the main station  $C$ . The stations  $B$  retransmit the information received to  $C$ .

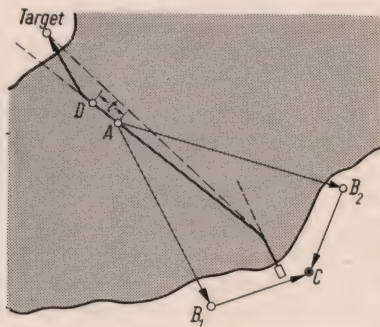


Fig. 8. Remote course and range correction of the V-1

From the three time intervals (missile to the three stations), the flying time, and the position of the stations with respect to each other, the location and mean velocity of the V-1 are calculated, and from the results a new course and a corrected distance to the target are determined for a point  $D$ , which is sufficiently far from  $A$  to have time enough for the necessary calculations to be performed. After the time  $t$  has elapsed, the corrections are transmitted to the bomb. For this purpose a tele-command receiver was incorporated in the bomb.

This receiver used the coded information to set the new course and to calculate the distance to the target.

In addition to its organizational duties, the Institute was required to develop and build the tele-command receiver with a translator. In this connexion, we were aided by the development work done by us for the purpose of replacing the amplitude-modulation technique of the "STRASSBURG-KEHL" with pulse technique. The required commands were represented by pulse combinations, which the receiver recorded on a magnetic tape, from which they were again picked off and retranslated.

The Institute built the first prototypes and initiated some small scale production. The control method had first to be tested in an aeroplane carrying the transmitter and receiver. Station *C* was built in Ainring; the two stations *B* were built 150 km away and were ready for operation. On Easter day 1945 working conditions became so bad that the flight tests could no longer be made.

## 7. FURTHER DEVELOPMENTS

### 7.1. Transmitting Device Using Ultraviolet Light

In the last year of the war small groups, displaced by the vicissitudes of war, joined the Institute. They brought their own problems and personnel with them. I shall mention only one of such groups, which had developed a transmitting device based on the use of ultraviolet light. The transmitter emitted very short, but very intense, light flashes which were directed to the receiver by a mirror. The receiver picked them up with the aid of a light-sensitive cell. When the sun did not shine on the cell, it was always possible to cover distances which were adequate for all the control operations required. Because of its excellent directional characteristics, the method was not subject to interference in practice.

### 7.2. Gyro-Stabilized Binoculars for Bomb-Aimers

It might further be remarked that an attempt was made to stabilize binoculars for bomb-aimers by means of gyroscopes. Because of the limited resolution of the eye, it was necessary to keep the flying bomb visible to the naked eye by pyrotechnic means. To eliminate this, and at the same time to make the target appear larger, the bomb-aimer had to use a pair of binoculars. The jerky motions and vibrations of the plane made this extremely difficult, and therefore gyro stabilization had to be used. The binoculars were prevented from rotating by means of two small gyroscopes and servo motors. The arrangement was relatively light, and could, moreover, be supported on the bomb-aimer's shoulder. Flight tests proved, however, that in the case of jerks the binocular-head-shoulder arrangement was not sufficiently rigid and produced blurred pictures. We were looking for another design which had to be particularly light and had to be worn in the manner of a pair of goggles. A satisfactory solution was not, however, found.

### 7.3. Theoretical Work on Target-Seeking Devices

In closing I should like to say that in the last months of the war considerable effort was directed to the "MADRID" project. The "MADRID" was a target-

seeking device based on infrared rays. It was to be used in anti-aircraft and night-fighter rockets against bombers. The results have not yet been covered by a report.

## 8. ACKNOWLEDGMENTS

I am glad that the writing of these works gives me the opportunity to thank those gentlemen, who have worked with me on this line: Prof. LEISEGANG, Dr. BACH, Dr. GLASER, Dr. SEEBACH, Dr. SCHEDLING, Dipl.-Ing. ERZ, Dipl.-Ing. SCHLÖGL, Ing. STAIGER, Ing. SCHREINER, and many others.

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## DISCUSSION

Question (anonymous): Was the accuracy of the components used in your simulators high enough to give a smooth indication?

Prof. Dr. FISCHEL: Yes, it was, because we used amplifiers between some devices. However, we had to overcome some trouble with the roughness of the test floor on which the different cars ran in the glider-bomb simulator. By using modern components, I would have built the simulator differently from the way I did it fourteen years ago.

Prof. Dr. WAGNER: For the guidance of the Hs 293, we were interested in a telescope that could be used to give an enlarged view of the target area aboard an aircraft. First we tried to attach a small telescope directly to the head of an observer, but we were very surprised to find that this made guidance more

difficult, even when used with the simulator on the ground. Obviously, the small and practically unnoticeable motions of the head confused the operator so that he could no longer correctly observe the motions of the (simulated) missile relative to the target, in spite of these two images being so close to each other. The final telescope which was successfully flight tested, consisted of the small ZEISS TURMON telescope stabilized by directly attaching it to a free gyro, with foam rubber inserted between. The field-of-view of the telescope was directed by a second observer. This rather elaborate two-man method was needed for the Hs 293. It is quite possible that a simpler method would be sufficient if the guidance system were designed from the beginning for use with a sight.

Prof. Dr. FISCHEL: I only have to remark that, when we started it, we considered the problem not to be as difficult as it turned out. After a while, we found that the gyros were well able to suppress turning motions, but we had no success with supporting the binocular to avoid translational motions relative to the eyes, and we found out that the human body is very elastic.

Admiral FAHRNEY (Philadelphia): In our use of television for guidance in America, we had great success when working against ship or land targets, with the assault drone approaching its target at a set altitude. However, when we attempted to pick up an air target we had great difficulty in picking it up and finally gave up the project. What success did you have, Dr. FISCHEL, with runs on air targets?

Prof. Dr. FISCHEL: Since our Institute made only test runs on ground targets, I would like to turn this question over to Prof. WAGNER, who might have some experience in this specific field.

Prof. Dr. WAGNER: The test flights with the Hs 293 D, which was guided by means of television, were not very successful. I was the test pilot for most of these launchings, and it was obvious that I had insufficient practice. Each launching gives only a few seconds of pertinent training, and we had no training device. We were too overloaded with work to develop a trainer ourselves, and we gave a contract for the development of a trainer to ZEISS-IKON but it was never finished. There was little emphasis on the use of television for guidance, because of the limited quality of the television picture.

# SPOILER CONTROL OF MISSILES

GÜNTER ERNST \*

## 1. INTRODUCTION

In reporting upon work done ten years ago, details may perhaps be forgotten but this is certainly to the advantage of a more complete general survey. I shall begin by mentioning Dr. MAX KRAMER, who was perhaps the first to recognize the value of spoilers and to make use of them. This report mainly is based on the development of the X-devices in the time the author worked together with Dr. KRAMER <sup>1,2</sup>. This paper is divided into two parts. First a coherent summary will be given of the effectiveness of the spoilers, employing an approximate method. In the second part the simplification in flight control resulting from the application of spoilers to guided missiles is discussed.

## 2. THE SPOILER AND ITS OPERATION

In England, about 1920, the use of spoilers was suggested as a remedy for the undesirable yawing moment of standard ailerons. In 1930 in America this concept was revived and in 1942 led to the development of a suitable lateral flight control system <sup>3</sup>. In Germany, the spoiler was especially applied to guided missiles such as Hs 293 C, FRITZ X, X-4 and X-7.

### 2.1. Definition of the Spoiler

Besides the word spoiler the expressions interruptor, interceptor or deflector have come into general use. All these terms signify a device to produce an aerodynamic effect by a burbled flow. In particular, the terms interruptor, spoiler or interceptor mean devices which produce a control effect by means of a change in lift opposed to the protruding direction of the control surface. The term deflector means a device which produces an effect in the direction of protrusion. In the following we shall consider only the use of spoilers for obtaining a control effect.

### 2.2. Spoiler in Two Dimensional Flow

For the start of our investigation we shall consider a two dimensional incompressible flow system.

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### 2.2.1. Spoiler on a Flat Plate

The simplest case we examine is a spoiler on the trailing edge of a flat plate (Fig. 1 a).

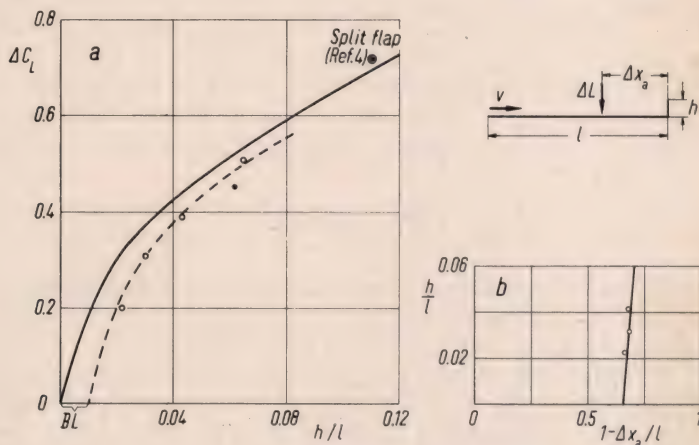


Fig. 1. Spoiler at the trailing edge of a flat-plate

a) Effectiveness of the spoiler  $\Delta C_L$  vs. height of the spoiler  $h/l$ ; cf. Eqs. (1) and (2)  
BL = Boundary layer

b) Line of action of the spoiler effect:  $\frac{\Delta x_a}{l} = \frac{1}{3} \frac{1}{1 + (1/2)\sqrt{h/l}} \approx \frac{1}{3}$

The spoiler induces a ram in front of itself, and a dead water area behind. This area is separated by discontinuity surfaces from the remainder of the flow. Reflecting the spoiler and its flow pattern on to the plate and considering this pattern, it appears to be similar to the flow around a static plate which has been treated theoretically by KIRCHHOFF by means of the potential function. Disregarding the changes of the flow at the leading edge of the flat plate, the KIRCHHOFF potential for the static plate certainly is a first approximation to the flow around a spoiler located at the trailing edge of a flat plate, in which the height of the spoiler is half that of the static plate. The streamline which passes through the stagnation point in the KIRCHHOFF flow pattern, represents, to a certain extent, the flat plate. To this approximation, the distribution of velocity and pressure along the plate due to the ram ahead of the spoiler is given.

By integrating the pressure distribution along the chord of plate and developing the expression for small heights of spoilers ( $h/l \leq 0.1$ ) we obtain the change of lift due to the spoiler:

$$(1) \quad \Delta C_L = 2 \sqrt{\frac{8}{4 + \pi} \frac{b}{l}} + \dots$$

$\Delta C_L$  is related to the dynamic head and to the surface composed of the chord of unit span ( $\Delta C_L = \Delta L/q l$ ). The factor  $8/(4 + \pi)$  in the square root of (1) is the

well known scale factor of the KIRCHHOFF representation. A practical rule of thumb for the effectiveness of the spoiler is:

$$(2) \quad \Delta C_L \approx 2 \sqrt{b/l}.$$

Fig. 1b shows the comparison of this result with old, unpublished tests. The agreement is good if the boundary layer is taken into account, i. e. if the height of the spoiler is reduced by the displacement thickness of the boundary layer. An additional test point is plotted showing the effect of a split flap on an airfoil with a plane suction side<sup>4</sup>. In this case also the above rule gives the correct values.

Naturally the centre of pressure, i. e. the line of action of spoiler effect, can be calculated from the pressure distribution. The result is that the line of action of the spoiler effect is at about a third of the plate chord in front of the spoiler.

The measurements shown in Fig. 1c demonstrate this fact quite well. So far a spoiler has been considered to be at a fixed point (trailing edge of the plate), but with variable height. Now the effects of the position of the spoiler in the airfoil chord will be discussed.

A spoiler located at a distance  $x_s$  behind the leading edge causes, in addition the ram in front, a vacuum behind itself. To use the results of the KIRCHHOFF calculations which presuppose that there is no vacuum, it is assumed now that the vacuum behind the spoiler does not appreciably change the velocity distribution ahead of it. Furthermore, a constant mean vacuum  $P_u$  behind the spoiler will be used in the calculations.

Thus the spoiler effect is composed of two parts. The component due to the ram is reduced first by the shorter effective plate chord, secondly because the reference area in this case is not the length ahead of the spoiler, but, as usual, the total plate chord. The vacuum also reduces the effectiveness; so the following expression for the spoiler effect is obtained:

$$(3) \quad \Delta C_L = \frac{\Delta L}{q l} \approx 2 \sqrt{\frac{8}{4 + \pi} \frac{b}{l} \frac{x_s}{l} - \frac{P_u}{q} \left(1 - \frac{x_s}{l}\right)}.$$

This result is shown in Fig. 2; the dashed line indicates the part due to the ram, whereas the full line represents the total effect. At a certain position of the spoiler a reverse effect occurs. It should be mentioned that the position of the spoiler at which the reverse effect occurs, is dependent on the height of the spoiler.

The vacuum can be reduced by supplying adequate air to the area behind the spoiler, thus increasing its effectiveness. This has been done in the American design of "plug-type-spoilers", which provides a slit on the pressure side of the airfoil for supplying air to the vacuum area on the suction side. Air can also be supplied from the ram area in front of the spoiler, e. g. using a toothed spoiler, as was done for the air-to-air missiles X-4.

### 2.2.2. Spoiler on an Airfoil

Now we consider a spoiler on an airfoil in two dimensional flow. In order to get an estimate of the control effect of a spoiler on an airfoil, the velocity potential of the airfoil without the spoiler is superposed on the KIRCHHOFF potential. Representing the pressure distribution on the airfoil by a TAYLOR

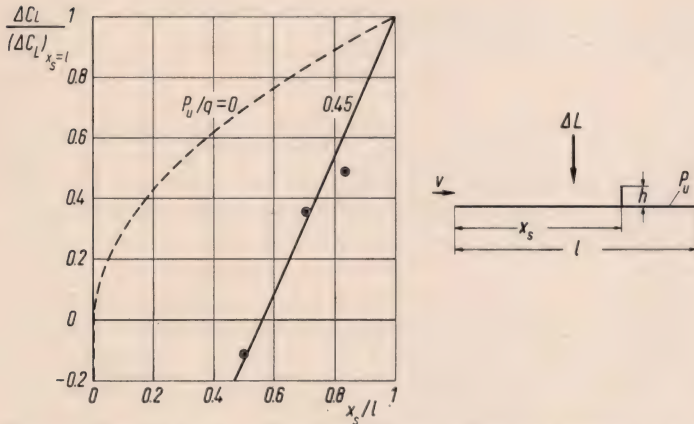


Fig. 2. The variation of the effectiveness of a spoiler with its location on a flat-plate

series with its origin at the spoiler, the first approximation of the spoiler effect, resulting from the superposition of the two velocity potentials, is given by the expression:

$$(4) \quad \Delta C_L \approx \Delta C_{L, \text{plate}} [1 - (p/q)_{x_s} + \dots];$$

$(p/q)_{x_s}$  is the local pressure on the airfoil, without the spoiler, at the spoiler position. The effectiveness of a spoiler on the suction side of an airfoil would be greater than that of a spoiler on a flat plate, because  $(p/q)_{x_s} < 0$ , while on the pressure side the effectiveness would be reduced [ $(p/q)_{x_s} > 0$ ], so that the spoiler acts in the sense of a deflector.

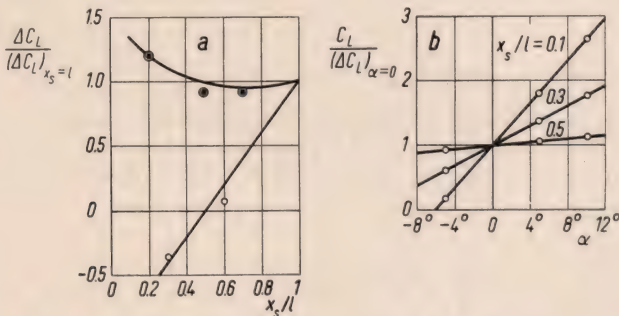


Fig. 3. Effectiveness of a spoiler on a profile

a) Measurements of an airfoil NACA 23012

b) Measurements of an airfoil Clark Y

Fig. 3a shows a comparison with some measurements<sup>5</sup>. From the theory the spoiler effect must vary linearly with the angle of attack of the airfoil. This is confirmed by Fig. 3b<sup>6</sup>.

The question of whether the spoiler is more effective at the trailing edge or in the neighborhood of the leading edge of an airfoil should now be answered. For a thin airfoil at a small angle of attack, the spoiler is most effective at the

trailing edge. For a thicker airfoil, the negative influence of the vacuum behind the spoiler could be compensated or overcome by the increased velocity.

The WAGNER bar control surface of the Hs 293 C acts like a spoiler at the trailing edge of an airfoil. The recognition of the fact that the increased speed increases the effectiveness of the spoiler incited the development of the airfoil step, as used, for example, in the devices X-3 and X-4. It was recognized that it is necessary to increase the local velocity of an airfoil at that place where the spoiler is located. Such a requirement can be satisfied by a suitable step in the shape of an airfoil, in which case a further increase of effectiveness occurs because of the decreasing thickness of the boundary layer. The influence of the step is especially effective for small spoiler heights, so that a doubling of effect can be obtained if the height of the step and that of the spoiler are of about the same order. It may be possible to gain a change of lift  $\Delta C_L$  of about 0.4 to 0.5 with a spoiler whose height is approximately 2% of the airfoil chord, located on a step of the same height.

### 2.3. Spoiler in Three Dimensional Flow

Previously a spoiler in a two dimensional incompressible flow has been discussed. To get an estimate of the effect of a spoiler on a wing, i. e. in three dimensional flow, it is convenient to replace its influence by an equivalent change of the angle of attack. Then we can apply the well known method for calculating the lift distribution along the wing span and we get the following formulation:

$$(5) \quad \Delta C_L = \Delta C_{L\infty} \frac{C_L'}{2\pi}.$$

It will be seen that the effect of a spoiler is proportional to the local lift distribution along a wing without a spoiler. Fig. 4 shows good agreement with measurements. This result has also been obtained elsewhere a short time ago, e. g. in France<sup>7</sup> and America<sup>8</sup> and has been confirmed by more recent measurements.

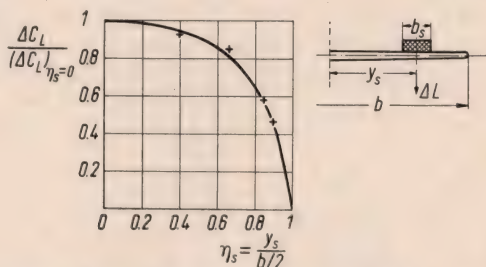


Fig. 4. The variation of the effectiveness of a spoiler with its spanwise location

Assuming an elliptical lift distribution, the effect of a spoiler extending over the total wing span is

$$(6) \quad \Delta C_L = \frac{\Delta L}{q b l} = \Delta C_{L\infty} \frac{1}{1 + 2/\lambda},$$

where  $\lambda$  is the aspect ratio. For a spoiler used as an aileron, we get an optimum position at 70% of half wing span, assuming elliptical lift distribution. In this case the spoiler has a span of 30% of wing span.

The realisation that the spoiler is more effective in two dimensional flow led, in the X-devices, to the principle of the arrangement of spoilers between end-plates. This use is apparent in the devices FRITZ X and X-7. In the X-4, the endplate is indicated by a reinforcing seam in the airfoil, but the end-plate effect is obtained by the sharp external edge which shifts the vortex farther outward.

## 2.4. Spoiler in Compressible Flow

To get an estimation of spoiler effect for changes in MACH number in the subsonic range, we can apply the PRANDTL rule, e. g. in the GÖTHERT form.

It was shown in paragraph 2.2.2 that the effect of a spoiler on an airfoil is determined by its height and its position, as well as by the velocity increase on the airfoil. In order to make an estimate on the basis of PRANDTL rule, an exact knowledge of the pressure distribution is necessary. However, we can consider two limiting cases which are easier to estimate. If the velocity increase is unimportant, that is if the spoiler is located on a thin airfoil or in the neighborhood of a trailing edge, we obtain:

$$(7) \quad \frac{\Delta C_L}{(\Delta C_L)_{M=0}} = \frac{1}{\sqrt{1 - M^2}}.$$

If, however, the velocity increase effectively controls the spoiler effect, e. g. for a step on airfoil, we obtain:

$$(8) \quad \frac{\Delta C_L}{(\Delta C_L)_{M=0}} = \frac{1}{1 - M^2}.$$

Fig. 5 compares the estimates (7) and (8) with measurements, and the previous considerations are confirmed. Since the PRANDTL rule is valid only for small incremental velocities, the decrease of effect naturally cannot be got beyond a certain MACH number.

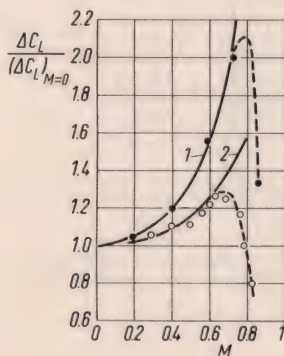


Fig. 5. The variation of the effectiveness of a spoiler with Mach number

1 = Airfoil with step, cf. Eq. (8)

2 = Airfoil without velocity increase, cf. Eq. (7)

● = Measurements, Ref. 1

○ = Measurements, Ref. 9

To some extent, the spoiler is effective far into the transonic range, as has been shown by tests on the X-3. A maximum MACH number of 0.98 was employed. Our experience with spoilers are limited to the subsonic range, but the spoiler should also be effective in the supersonic range (W. SEIBOLD in the last WGL Meeting <sup>10</sup>).

### 3. APPLICATION OF THE SPOILER TO FLYING BODIES

The spoiler has the important property of having a small power requirement because only the inertia forces must be overcome, since the aerodynamic forces are negligible if properly constructed. For usual aircraft, the important difficulty in the application of spoilers is, however, the operational delay.

During the protrusion of a spoiler the burbled flow must have been developed for a certain length in order to produce an effect. This causes a certain time lag, which essentially is proportional to the wing chord and inversely proportional to the airspeed. The disadvantage of the time lag can be reduced to a very small value on guided missiles, because the wing chord is small and the airspeed is high, for instance the time lag for the FRITZ X device was of the order of a few thousandths of a second.

#### 3.1. Construction of the Flight Control System

A spoiler is a near inertialess control surface with a small power requirement, which produces important technical simplifications in guided missiles, since it favorably affects the total space requirement, weight, supply, and reliability in service of the control system.

To guarantee a constant effectiveness of the controls for positive and negative angles of attack, symmetrical airfoils with rear position of the maximum thickness were chosen for the X-devices, and the spoiler with its protrusion mechanism was placed at the point of largest thickness. The spoilers of both sides of the airfoil were connected together, both protruding halfway in the mid-position. This fundamental construction of the controls was retained for all the X-devices, but the shape of the airfoil (step) and the height of the spoiler were adapted to the requirements of each device.

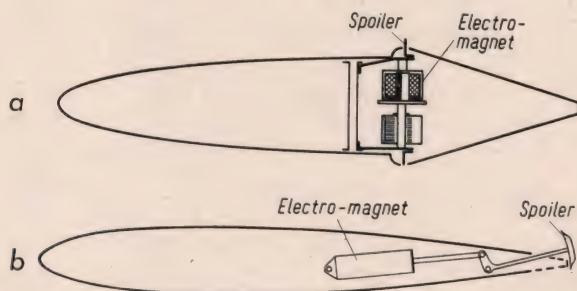


Fig. 6. Example of spoiler arrangement  
a) Control of device X-3      b) Bar control of Hs 293 c

Fig. 6a shows, as an example, a cross section of the controls of the X-3, which had a power requirement of about 2 Watts. The structure of WAGNER's bar control surface in which the spoiler was arranged at the trailing edge of the wing, is shown fundamentally in Fig. 6b.

### 3.2. Method of Steering

From the fact that the spoiler can be thought of as an inertialess control, important simplifications follow for the operation of steering. Without increased expenditure in a "yes-no" steering system, any desired value of control effect can be obtained from a suitable temporal distribution of full movements of the control surface. It is only necessary to make the frequency of the periodic change of deflection of the control surface sufficiently large, so that the missile, due to its own small characteristic frequency, correctly averages the short-spaced changes of the full movements of the control surface. The modulation frequency of the spoiler was 5 c/sec for the missile FRITZ X, whose flying weight was 1,500 kg, and the natural frequency of the missile at its highest flight velocities was 1 c/sec.

The small power requirement made the use of a simple slot relay on the receiver output possible. In this way, for the wire transmission of the X-4, it was possible for the SIEMENS relays to assume the task of relaying and the task of signal discrimination at the same time, i. e., the selection of the control commands.

### 3.3. Application of Spinning Flight Control

A further simplification could be obtained by the application of spinning flight control. Considering the FRITZ X device, all 6 control surfaces are

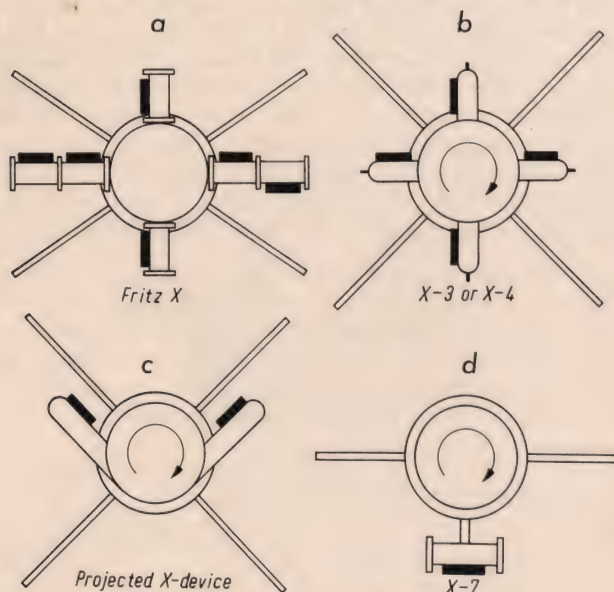


Fig. 7. Development of the X-missiles

located together in the tail section. Always two control surfaces correspond to one control axis (Fig. 7a). The elevator and the rudder are controlled by the commands of the guiding system, whilst the ailerons are set into operation by an artificial stabilization-system which guarantees the position of the missile. For missiles with axially-symmetric operative wing arrangements, such as cross wings, the stabilization about the roll axis can be avoided, and instead stabilization of the control direction or rotational flight control could be used. If the missile rotates at constant velocity around its roll axis, it is sufficient to operate the suitable control surface at each instant to get the desired command result. Such a control system does not need ailerons, but rather a gyroscope, equipped with a control selector which would correctly select the control surface corresponding to the required steering direction. For satisfactory averaging of the commands, the body must rotate fast enough. This could be insured by a twisted angle of attack of the wings. Fig. 7b shows such an arrangement of the control surfaces, in which the number, and therefore the power consumption, has been reduced from 6 to 4.

By combining the symmetrically arranged control surfaces, the number can be reduced to 2 as shown in Fig. 7c, for a projected X-device. The spinning flight control permits an increase of the construction tolerances, for the average of such errors produces no deflections from the path.

The guided anti-tank rocket X-7 represents a still further improvement. A steering effect about two axes is attained by a single control member. The device is a rotating mono-wing in which the altitude and lateral commands are chronologically put into operation by a control selector stabilized by a gyroscope. If the wing is in the horizontal position ( $\pm 45^\circ$ ), the elevator would be set in operation. After a quarter turn, when the wing is in a vertical position ( $\pm 45^\circ$ ), the rudder is set into action, and the device flies without the lift of the wing in vertical plane. The rotation velocity must be chosen with great care to get small path deflections. The natural frequency of the control surface must be considerably higher in order to take the mean of the command values. The use of a spoiler as a control surface permits the realisation of these requirements. The guided anti-tank rocket X-7, with a total weight of 9 kg was probably the device which had the minimum of power for control.

#### 4. SUMMARY

The spoiler is a control surface whose effectiveness can be predicted without trouble. In the case of guided missiles, the aerodynamically determined time lag is practically of no importance, so that the important property of an inertialess control can be used easily. By utilization of this property, the continuous control of a missile can be simplified so that the control consists of a single control surface with a "yes-no" operating system and a control selector stabilized by a gyroscope.

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## DISCUSSION

Prof. Dr. QUICK (Aachen): Herr ERNST has given us a summary of an important detail problem concerning the development of the X-devices. As far as I know, the spoiler has also undergone several tests in the supersonic range. Are any results available on the influence of the shape of the spoiler in the supersonic range? What has to be especially considered to avoid a reverse of the effect desired?

Ing. ERNST: The efficiency of a spoiler depends on the height and on the form of the shape step. There is a maximum height of the profile step corresponding to each MACH number; an increase of the MACH number is followed by a decrease of the step height.

When the spoiler is withdrawn, it has to be secured, so that the profile step itself does not cut off the flow. This means that the transition from the profile step to the back ridge would have to decrease with incidence, as the MACH number increases. The profile length would therefore increase, if this incidence could be pushed to a back ridge thickness of zero. Since the under pressure area increases, the spoiler efficiency will be reduced. To avoid this development, we agreed to admit a definite dimension of the back ridge, because the drag problems are of minor importance with guided missiles.

To get a sufficient spoiler efficiency within a large MACH number range, the span was combined with a variable step height and back ridge thickness in such a manner that larger step heights correspond to smaller back ridge dimensions, and vice versa. The basic idea was that at smaller MACH numbers the spoiler efficiency was to lie within the range of the largest step heights. With increasing

MACH number, this range of efficiency was to move towards the range of the small step heights. The tests (see Ref. 1) proved this idea to be right up to a MACH number of about 0.9. But no testing was done within the supersonic range.

It may be quoted here, that delta shaping of the spoiler showed no results. This is understandable, since the lateral flow reduces the ram effect of the spoiler and therefore its efficiency.

Dr. KRAMER (Pacific Palisades, Calif.): We did not have the means to perform tests within the supersonic range.

# THE AERODYNAMIC DEVELOPMENT OF THE V-2

HERMANN H. KURZWEG \*

## 1. INTRODUCTION

The design of V-2 type rockets is greatly influenced by the considerable aerodynamic forces and temperatures to which high-speed vehicles are exposed during their flight. Shape, structure, and performance depend to a great extent on the knowledge of the aerodynamic and thermodynamic conditions encountered in supersonic flight, and these were not well-known at the beginning of the long-range rocket development. Adequate experience was not available, since a vehicle of the contemplated size and speed of the A-4 had never before been built. Systematic studies of drag, stability, control, and aerodynamic heating of the supersonic body had to be initiated before long-range rockets could be successfully launched.

### 1.1. First Aerodynamic Contacts

This situation was recognized in the early stages of rocket development by General DORNBERGER and Dr. VON BRAUN, after the launching of their first large liquid rockets "MAX" and "MORITZ" before 1936. These rockets carried heavy gyros internally to obtain stability. However, because the aerodynamic forces were not taken into account in their design, they failed to stabilize the vehicles shortly after take-off.

They therefore contacted, the TECHNISCHE HOCHSCHULE at Aachen, where Professor WIESELSBERGER and Dr. R. HERMANN had constructed a small  $10 \times 10$  cm supersonic wind tunnel<sup>1</sup> in which aerodynamic tests on very small scale models could be performed.

In this tunnel, the first supersonic aerodynamic data for the newly-planned rockets were obtained and, in particular, the shape of the test rocket A-3, "A" standing for "Aggregat", was developed. The A-3 was a large fin-stabilized liquid oxygen-alcohol rocket of approximately 80 cm diameter and 8-calibre length. For the tail surfaces, very slender fins were selected which extended several calibres behind the base. At their aft-end they carried an antenna ring which gave them structural strength. The tunnel tests indicated that such tail structures could stabilize the rocket in supersonic flight.

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## 1.2. Aerodynamic Department at Peenemünde

The A-3 was supposed to be the forerunner of the A-4, which much later became known as a weapon by the newspaper name of V-2. Since it was clear that the aerodynamic development of the much larger and faster A-4 could not be done satisfactorily in the small Aachen tunnel, in 1936 General DORNBERGER asked Dr. HERMANN to build a larger  $40 \times 40$  cm supersonic wind tunnel at Peenemünde. This was the beginning of the activities of the Aerodynamic Department, which originally consisted of three divisions: Mechanical Design under H. GESSNER, Electrical Engineering under H. RAMM, and a Research Division run by the writer in conjunction with his job as Deputy to R. HERMANN, the Head of the Department. It was the responsibility of this Research Division ("Forschungslabor"), to furnish the aerodynamic and thermodynamic information which was needed by the mechanical designers. The main work was done between 1937 and 1945, first for the A-4 and its test vehicle A-5 (which replaced the A-3), and later on for the anti-aircraft rocket "WASSERFALL"<sup>2</sup>, the arrow-shells "Peenemünder Pfeilgeschosse", and several projects for the German Army and Air Force.

## 1.3. Peenemünde Supersonic Wind Tunnels

The main tool of the Department was the  $40 \times 40$  cm wind tunnel, an improved and enlarged version of the Aachen tunnel. It was mechanically designed by H. GESSNER and his group, and it was built at Peenemünde between 1937 and 1939. Blowing times of 18 to 20 sec were obtained by evacuating a  $1,000 \text{ m}^3$  sphere with a 1,000 HP system of three twin-vacuum pumps. Since it was a free-jet tunnel, the pressure in the test chamber had to be controlled by setting an adjustable diffuser, to avoid strong compression or expansion waves originating at the nozzle exit. A three-component electromagnetic balance, located within the test chamber, was used for force measurements. The model was mounted on a sting whose angle of attack could be varied by remote control. Originally, only  $M = 1.5$ ,  $M = 1.75$ , and  $M = 2.25$  nozzles were built, using the PRANDTL-BUSEMANN graphical design technique. The flow, with MACH number variations along the axis of from 1% to 3%, was considered to be sufficiently smooth in spite of the manufacturing imperfections which showed up as in the schlieren pictures a mass of MACH lines superimposed on the model wave pattern. The nozzles could be corrected by measuring the MACH angle and it was later found that, at higher MACH numbers and with smoother nozzles, the absence of these lines presented a real problem for the nozzle correction since the pressure differences on the flow axis could not be so easily traced to their origin at the wall. The optical bench with two optically corrected 50-cm mirrors, each with a focal length of 5 m, was the largest, most sensitive schlieren equipment at that time. A large MACH-ZEHNDER interferometer, with 25-cm plates of perfect quality made by C. ZEISS, Jena, was installed at the  $40 \times 40$  cm test section.

Later, when the MACH 3.3 nozzle was built, no satisfactory test rhombus could be obtained for a long time, because the downstream part was limited by strong compression waves. This compression could not be avoided, even with the diffuser fully open, but finally the reason was found to be leakage of the supposedly sealed test section.

R. LEHNERT's plot of chamber pressure against the leakage area, obtained by step-by-step opening of the test section windows, showed that even in the case of "everything" closed, some area equivalent to 15 cm<sup>2</sup> was still open. The problem was solved by sealing fine gaps and screw holes. This was a step in the development which today seems very trivial, but which, at that time, caused terrific headaches, and was a discouraging obstacle on the way to the necessary higher wind-tunnel MACH numbers. By 1943 the Aerodynamic Department operated two almost identical 40 × 40 cm test sections up to MACH 4.4, one 18 × 18 cm tunnel, and one approximately 10 × 25 cm vertical test section for the development of nozzles and diffusers for higher MACH numbers.

During 1943, the last year at Peenemünde, S. ERDMANN, in order to check calculations on air liquefaction, designed a reduced cross-section MACH 8 nozzle, with a high-pressure air supply, for the 40 × 40 cm tunnel. According to C. WAGNER, Darmstadt<sup>3</sup>, liquefaction should occur at MACH numbers above five. These investigations were made for the planning of a new large 1 × 1 m, MACH 10 hypersonic tunnel in the Bavarian Alps in which the aerodynamic knowledge for the rockets of the future, for instance the A-10, could be advanced. The tests of the MACH 8 nozzle were, however, discontinued because of the air raids and the relocation of the Department, with all the tunnels, to Kochel, Bavaria, where the activity was renewed under the new name WASSERBAU-VERSUCHSANSTALT, Kochel.

#### 1.4. Free-Flight Small Test Vehicles

The wind-tunnel results were obtained on relatively small models — up to 50 times smaller than the full-scale rockets. Furthermore, the small models had to be mounted on relatively big stings and possibly unknown characteristics of the air flow could produce errors in the measurements. Therefore, in order to confirm the results as much as possible on free-flight vehicles, large quantities of inexpensive 20- and 30-cm diameter model rockets were launched, and heavy solid models were dropped from aeroplanes.

The rocket models, WALTER rockets filled with hydrogen-peroxide, were built with a great variety of body shapes and fin configurations designed with more or less theoretical concepts, based mainly on previous subsonic wind-tunnel results. These free-flight investigations greatly advanced the understanding of the dynamic behaviour of the fin-stabilized rockets. These were the tests and observations on which SCHNELLER, Darmstadt<sup>4</sup>, based his calculations of the "lunar" motion, the resonance effect of pitch and roll, which for the first time gave a plausible explanation of the catastrophic "corkscrew" oscillations which prevented the test rockets from travelling through the speed of sound. According to SCHNELLER's theory, the uncontrolled rocket models begin to spin during the relatively slow acceleration due either to the asymmetry of the fins or, even with perfect symmetry, to rolling moments induced by side wind or pitch oscillations. Subsequent measurements in several subsonic wind tunnels (Darmstadt, Göttingen, and Friedrichshafen) led to a satisfactory explanation of this "lunar" motion, at which pitch and roll frequency are equal.

Because of this behaviour, no information in the transonic region could be obtained with these rockets, and so other free-flight tests had to be made. It was calculated that solid steel models of 20 cm diameter could, because of the low drag coefficient and high weight, attain supersonic speeds if dropped

from aeroplanes at 7,000 m altitude. A great number of such drops were made and, satisfactory observation was achieved after the initial difficulties of the test technique, using ASKANIA phototheodolites, had been eliminated. These bodies seemed to be perfectly stable and exceeded the speed of sound shortly before they reached the ground. The observation time of the critical region was, of course, very short and it was, and remained, somewhat doubtful if three to five degrees angle of attack could be measured on the very small theodolite pictures.

## 2. RESEARCH AND DEVELOPMENT WORK

### 2.1. Aerodynamic Data

#### 2.1.1. Available Basic Information by 1937

When the systematic aerodynamic work started in 1937, not much information was available on bodies at supersonic speeds. Scanty experimental data could be collected from ballistic investigations which, in general, were carried out on spin-stabilized projectiles. The behaviour of the spinning projectile had been treated mathematically during the past decades with gyroscopic equations from which a stability coefficient ( $\sigma > 1$ ) and a "Folgsamkeitsfaktor" (ability of the projectile axis to follow the trajectory tangent) had been deduced<sup>5</sup>. However, the components of the aerodynamic forces and moments and their complicated interactions, especially in supersonic flow, were practically unknown. Some data on the drag of spinning projectiles were obtained from full-scale firings, but normal forces, MAGNUS forces, and damping forces in supersonic flow were practically unexplored, experimentally, at that time.

Because of its low drag, the shape of the German 8-mm infantry bullet was originally selected for the first design of the A-3 and A-5. This was done because supersonic data on fin-stabilized missiles were even more scanty. Some people, probably influenced by remarks in the literature, were of the opinion that fin stabilization in supersonic flow is not possible. The observations were possibly based on the erratic trajectories of projectiles with fins of a diameter equal to or smaller than, that of the body, especially when they were fired from rifled guns. R. HERMANN's measurements on the small A-3 models in the 10-cm wind tunnel in Aachen showed, however, that even at a low supersonic MACH number, it is possible to force the centre of pressure behind the centre of gravity by feasible tail surfaces.

For theoretical work, the members of the "Forschungslabor" had to lean heavily on BUSEMANN's publications, in particular, on his Handbuch article on "Gasdynamics"<sup>6</sup> and the published talks by VON KÁRMÁN, PRANDTL, ACKERET, etc. given at the Volta Congress in Rome, 1935<sup>7</sup>. Also the article by VON KÁRMÁN and MOORE, "Resistance of Slender Bodies"<sup>8</sup> which had been published previously, was used to calculate pressure distribution of the A-4 and A-5 bodies using the method of sinks and sources. Later on during the war, W. TOLLMIE, Dresden, and R. SAUER, Aachen, developed their methods of characteristics for axisymmetric bodies and calculated for us the pressure distribution over the A-4 bare bodies. Fig. 1 gives an example of the results

obtained by the various methods, at zero angle of attack, as compared with the test results which will be described later in this report.

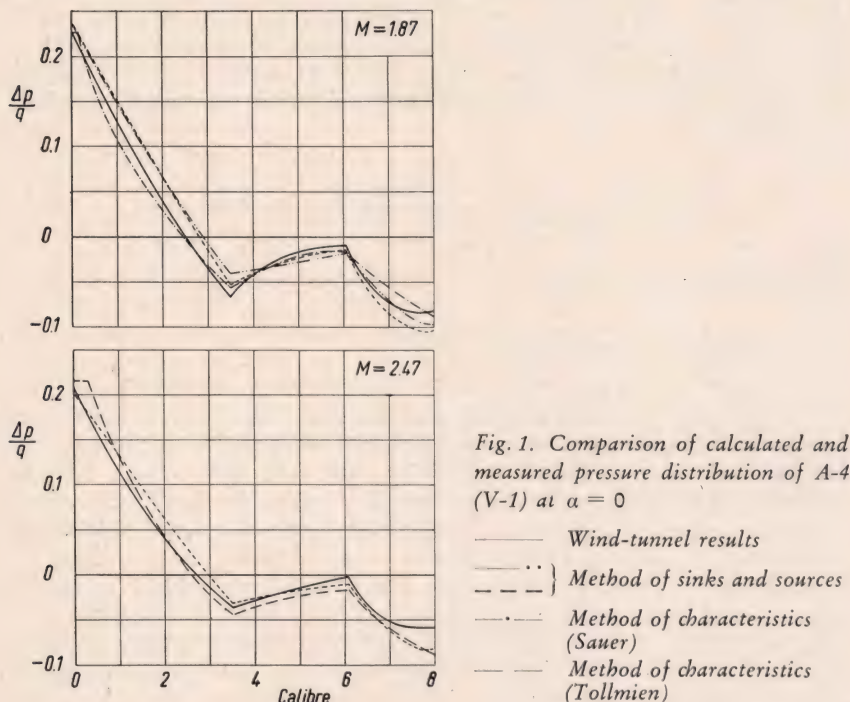


Fig. 1. Comparison of calculated and measured pressure distribution of A-4 (V-1) at  $\alpha = 0$

— Wind-tunnel results  
 - - - Method of sinks and sources  
 - · - Method of characteristics (Sauer)  
 — — — Method of characteristics (Tollmien)

It can be seen that these potential flow calculations gave a good over-all agreement with the experiment. The integration, however, of the tangential components over the whole body gave drag values which did not agree too well with the direct drag measurements. Only after friction and base drag had been estimated as correctly as possible, was satisfactory agreement obtained. However, from our present knowledge any good agreement between the results of theoretical calculations and of firing experiments and wind tunnels at that time must have been more or less accidental. This is because the variation of the base pressure with supersonic MACH and REYNOLDS numbers, e.g. as function of the boundary-layer characteristics with and without heat transfer, was unknown. Laminar friction had been calculated<sup>9</sup>, but turbulent boundary-layer theory did not exist. No studies of transition from laminar to turbulent supersonic boundary layer had been made.

A few measurements were made of the base pressure on cone-cylinder bodies, but only the MACH number was varied. According to these measurements, the base pressure ratio,  $p_b/p$ , where  $p$  is the undisturbed free-stream pressure, decreased with increasing MACH number to approximately  $p_b/p = 0.5$  at  $M = 2.5$  and increased then again at  $M = 3.3$  and  $M = 4.4$ . This increase indicated that, at very high MACH numbers, there would be practically no base suction. Years later, the explanation of these remarkable results was to be found in the influence of the boundary layer on the base pressure, which is

strongly REYNOLDS number dependent. With laminar boundary layer at constant MACH number, the base pressure increases with decreasing REYNOLDS number. The measurements were made under conditions where MACH and REYNOLDS number varied simultaneously.

Thus, the work of the Aerodynamic Department started practically in new territory but the first relatively primitive approaches were very soon replaced by a well-directed supersonic aerodynamic research.

### 2.1.2. *Start of Aerodynamic Development Work at Peenemünde*

The first basic aerodynamic work started at Peenemünde in 1937 on the A-5 rocket, the improved version of the A-3, which required entirely new tail surfaces since the fins with antenna ring were operationally and aerodynamically inadequate. A few A-3 rockets (practically the same body as A-5) were already under construction, and they were fired from the island Greifswalder Oie at the end of 1937. The failure of these rockets occurred due to the guidance system long before the jet, expanding with altitude, could have badly burned the fins and destroyed the antenna ring and the stabilization.

The first three fin designs for the A-5 were made by the author from rough estimates of the pressure distribution over the body and a few available subsonic normal force and centre of pressure data for flat plates of aspect ratios below one. These three designs carried the letters V-1, V-2, and V-3 according to the German "Vorschlag" (i.e. "proposal"). The V-1 was the fin which, with minor changes (denoted subsequently with letters such as V-IF, etc.), finally became the tail surface of the A-4. The relation between the later newspaper name V-2 for the A-4, and the original denotation of the fins is purely coincidental. This, by the way, was hard to explain to the first investigators after the war who thought V-3 was another secret weapon of the V-series. The denotation of the following tail surfaces was carried on, as far as I remember, to the number V-16 each with several variations with respect to thickness, leading and trailing edges, and root configurations.

The fin V-1 with a  $60^\circ$  sweep extended 2 calibres in front and a half of a calibre behind the body base. The fin V-2 was smaller and the V-3 was larger in area. A  $45^\circ$  sweep of the trailing edge was supposed to leave ample space for the expanding jet at high altitudes. The outer edges of the fins were divergent at approximately  $2^\circ$  to the body axis. The span was 2.2 calibre.

Because information was needed immediately and the Peenemünde tunnel had not been built, the first model with the three different fin configurations was carved, over the week-end, by the author from a 1-inch Peenemünde pine branch. The fins V-1, V-2 and V-3 were made from scrap hard rubber plates and the estimated centre of gravity was trimmed by inserting brass plugs in the model (Fig. 2).

The first tests were made by throwing this model into the wind from the roof of a two-story house, only to learn, of course, that the inertia term, in the differential equation governing this "oscillator", was much too large compared with the very small aerodynamic moment developed during the available falling time.

A more reliable test was carried out by suspending this model on a wire through the centre of gravity and allowing it to rotate freely, mounted on a car travelling at 100 km/h. From this "test" it appeared that the V-1 and V-3 were

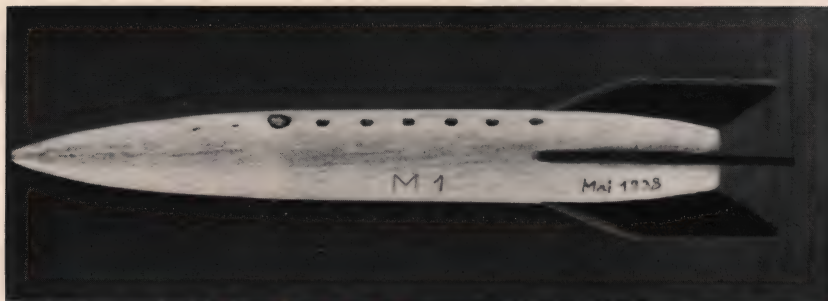


Fig. 2. First hand-carved model of the V-2

stable whereas the V-2 was unstable within a restricted region at angles up to about 5 or 10 degrees. At larger angles it was stable. It was a case of circular stability.

### 2.1.3. Results from Subsonic Wind Tunnels

Subsequently, systematic tests were carried out on well-made models in various subsonic wind tunnels in Germany. The subsonic ZEPPELIN wind tunnel in Friedrichshafen, under the direction of MAX SCHIRMER, contributed most to the understanding of the subsonic aerodynamic behaviour of the new rocket bodies. SCHIRMER, who had first-hand experience in the investigation of the similar ZEPPELIN bodies, made detailed pressure measurements over the whole surface. His demonstration models showing the three-dimensional pressure distribution on body and fins were a great help in allowing the mechanical rocket designers to "get the feel" for the aerodynamic loads on the weight-saving structure and skin. Tests were made on the finned body, and on the body alone, to study the interference problem. So, for the first time, the varying magnitude of the forces at all locations on the model and their moment arms could be visualized, stabilizing and destabilizing moments recognized, and centres of pressure as function of angle of attack determined from the individual components. Also, it was possible to obtain a better estimate of the necessary size and position of the fins.

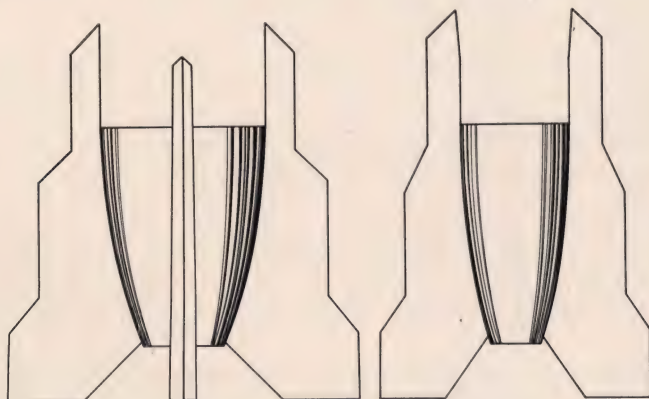


Fig. 3. Three shoulder fins

It was evident from these tests that the main aerodynamic net normal force (difference of pressure on both sides) was concentrated close to the shoulder of the fins. Therefore the stability was largely due to the location of the aerodynamic force close behind the centre of gravity, and cutting of the trailing fin tips did not appreciably affect the restoring moment. This fact led to the conclusion that a multishoulder fin in the form of a Christmas tree was a better stabilizing tail surface than a single-shouldered fin (Fig. 3). In this way, the one large shoulder force on a short moment arm was distributed over three smaller shoulders on successively larger moment arms. The tests showed that the stability could really be increased considerably with these fins, but the lift-drag ratio was not too good. These pressure measurements also showed clearly that the idea of letting the low-aspect ratio fins extend over the full length of the body could not lead to a well-stabilized body with a favourable lift-drag ratio in the speed range of interest at that time.

#### 2.1.4. Damping Investigations

The first dynamic measurements of aerodynamic damping coefficients were also obtained in the SCHIRMER tunnel with and without jet, on models about 2 m in length. They were free to oscillate in a horizontal plane about an axis through the centre of pressure. Compressed air was led into the jet model via a labyrinth seal, through the vertical spindle.

The problem of aerodynamic damping, which was of primary importance to VON BRAUN's guidance group, was intensively attacked by our Research Division and W. HEYBEY<sup>10</sup> worked out solutions with linear, quadratic, and cubic aerodynamic restoring moments in the modified differential equation

$$\Theta \ddot{a} + R \dot{a} + \bar{M} a = 0,$$

where the first term represents the inertial forces, the second term the damping, and the third term the restoring moment with linear character. In the course of these investigations, the damping coefficient  $R$  was transferred into a dimensionless damping coefficient  $C_d$  which was adaptable to similarity studies. In the form

$$C_d = 24 R / \rho v D L^3,$$

it was presented as a function of MACH and REYNOLDS number. This coefficient  $C_d$ , derived by using a very simple concept of a swinging flat plate, was very useful in the calculation of the full-scale dynamic behaviour of the rockets. Its magnitude is of the order of unity, whereas the coefficient  $R$  varies several orders of magnitude between wind-tunnel models and full-scale bodies.

The dynamic tests were later continued in the Peenemünde tunnel with a similar, but horizontal, support for the models, and valuable supersonic results up to  $M = 3.3$  were obtained. Fig. 4 shows a typical graph of the logarithmic decrements of two tests with and without jet. The test points can reasonably be connected by straight lines from which  $R$  and  $C_d$  can be calculated. It is

$$R = \frac{7}{3} \Theta \tan \varepsilon$$

where  $\Theta$  is the moment of inertia and  $\tan \varepsilon$  is the slope of the logarithmic decrement. The factor  $7/3$  is due to the cubic restoring moment; it is replaced by

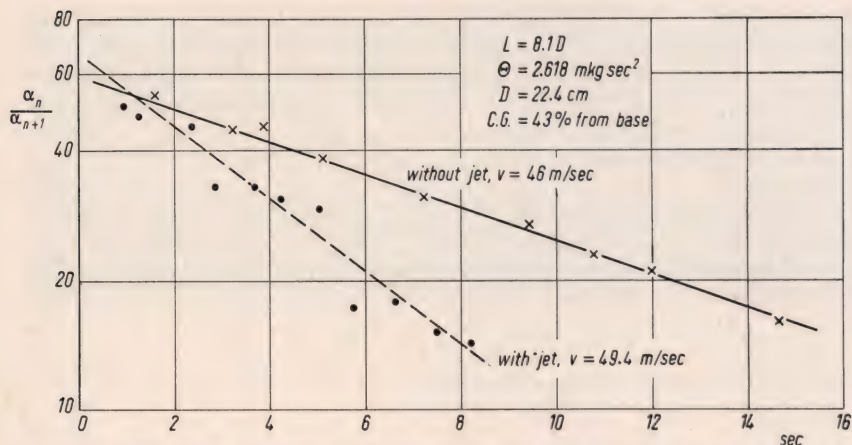


Fig. 4. Logarithmic decrements of oscillations with and without jet. Model A-5

the factor 2 if the moment is linear. According to these tests, the jet increases the damping coefficient by a factor of two.

Fig. 5 shows the measured damping coefficients as function of REYNOLDS number. The curves connect the data obtained in the subsonic tunnel at Friedrichshafen. Two supersonic points, each for  $M = 1.86$  and  $1.56$ , for two different axes of rotation (centre of gravity 43% and 38% respectively of the length from the base) are also plotted. They are close to the subsonic values for the corresponding REYNOLDS numbers.

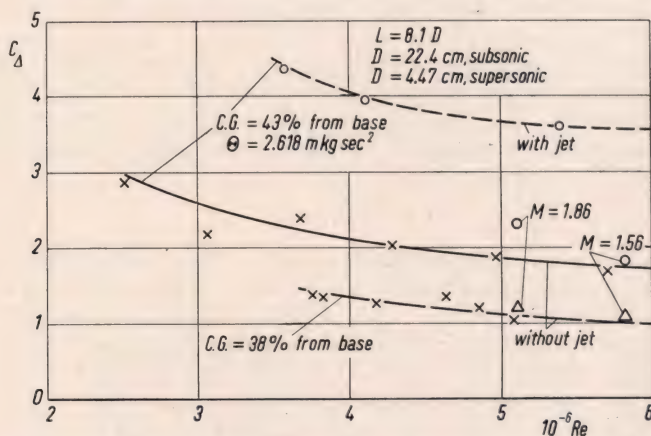


Fig. 5. Damping coefficients as function of Reynolds number. Model A-5

These oscillation tests, showing how the A-4 behaves when it re-enters the atmosphere, were one of our more spectacular demonstrations for visiting VIP's. The model, even when started at  $180^\circ$  angles of attack turned rapidly into the wind and stabilized itself with rapidly decreasing amplitudes. The wire mount, which allowed the model to swing the full  $360^\circ$ , was later replaced by a sting mount which permitted oscillations of only one or two degrees. The sting, whose

angle of attack could be varied by remote control, reached through the base into the hollow model to the pivot point which could be moved along the axis to allow for different c.g. positions. In this way no side holders or wires disturbed the flow, and very accurate measurements of the zero-moment positions of point, and circular, stable models could be made.

### 2.1.5. Force Measurements in Supersonic Flow

During the shakedown period (1939—1940) of the Peenemünde tunnel the first force measurements were made with the external electromagnetic three-component balance. Reliable data were finally obtained after the air dryer equipment had been installed. This was a system which R. HERMANN and G. EBER had worked out with the SILIKAGEL Company and resulted in a patent for air dryers for intermittent wind-tunnel operation.

R. LEHNERT's measurements of drag, lift, and centre of pressure can be considered to be the first reliable data on the A-5 body obtained in the Peenemünde tunnel (Fig. 6). Afterwards, the corresponding curves for the A-4, which was very slightly different in shape of body and fins, were obtained (Fig. 7). Later on systematic investigations were carried out to find practical bodies of minimum drag in viscous supersonic flow. Such studies were made not only to

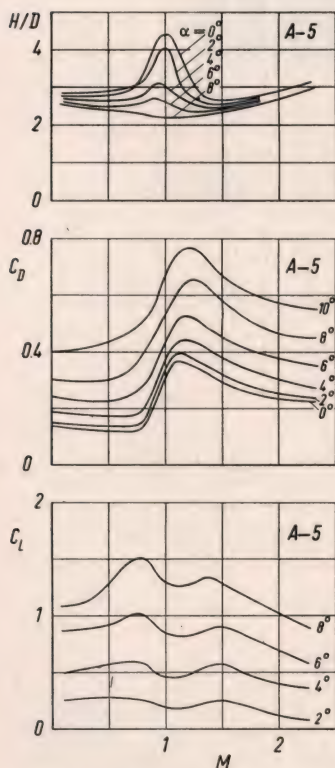


Fig. 6. Aerodynamic coefficients of the test rocket A-5

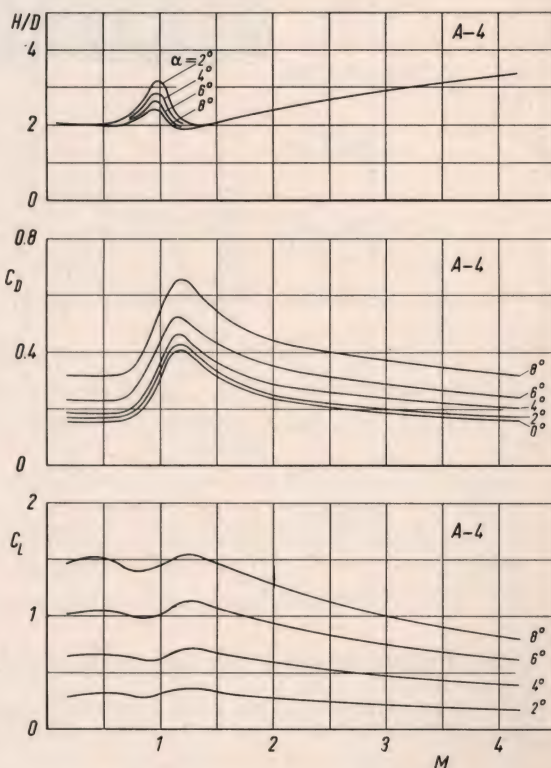


Fig. 7. Aerodynamic coefficients of the rocket A-4

find optimum drag bodies for rockets but also, more generally, to find shapes for undercalibre arrow-shells to be fired from guns by means of sabots. Both bodies require a flat base of a certain diameter, the rocket to accommodate the nozzle exits, and the projectile to carry its own mass during the acceleration without exceeding the strength of the material. Fig. 8 shows one of several of the body families investigated where systematic changes of nose angle, volume, length, position of largest cross section, and their respective combinations were made<sup>11</sup>. To overcome the difficulty of comparing the drag coefficient of geometrically different bodies, three different areas of reference were used: the largest cross section, the surface area, and the area computed from the volume to the two-third power. Unfortunately, there was never enough time and

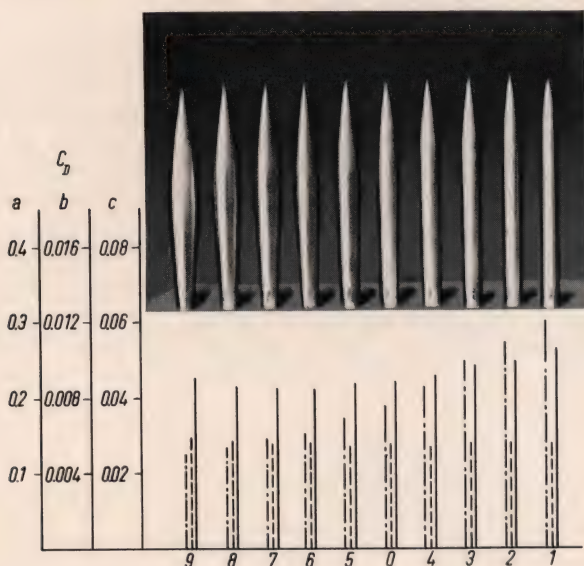


Fig. 8. Family of bodies for minimum drag investigation

a) Largest cross section

b) Surface area

c) Area computed from the volume to the two-third power

manpower to carry these systematic basic investigations to fully satisfactory results and conclusions. However, these few tests at only one MACH number ( $M = 2.5$ ) and one REYNOLDS number (which varied slightly with the model length), contributed useful material, especially for the development of the Peenemünde arrow-shells.

### 2.1.6. Pressure Measurements with "Half Models"

A progressive step in the continuous improvement of the test techniques was made by the introduction of the "Halbmodelle", which are models bisected in a plane of symmetry and mounted on a flat separation plate. S. ERDMANN<sup>12</sup> developed this method into a very powerful tool. Previously, the difficulty in obtaining, economically, pressure distribution data over the whole body

and fins was very great, because only a very few pressure tubes, which connected the local pressure holes with the outside manometer, could be led through the small mounting sting. With the model diameter not larger than 4 to 5 cm, the sting could carry only a few tubes and the diameter of the tubes could not be made too small, because of the increasing internal resistance and the restricted running time of the tunnel. The half model idea solved this problem perfectly. Up to 120 tubes were mounted in the model and brought through the plate on the opposite side where the mountings and a large number of tubes could not disturb the flow over the model itself. Checks for comparison on half-model data and those obtained from "full" models showed reasonable agreement, even at the junction line of plate and model, which was clearly within the boundary layer of the plate. With this method, the test time could be cut to less than one tenth of that required otherwise. More than 100,000 readings on the A-4 model, at about 10 subsonic and supersonic MACH numbers, could be made in about two weeks in a two-shift operation, with 20 people each reading manometers and taking notes. The evaluation of this tremendous amount of test data took, however, quite a time. The results are shown in Figs. 9 and 10.

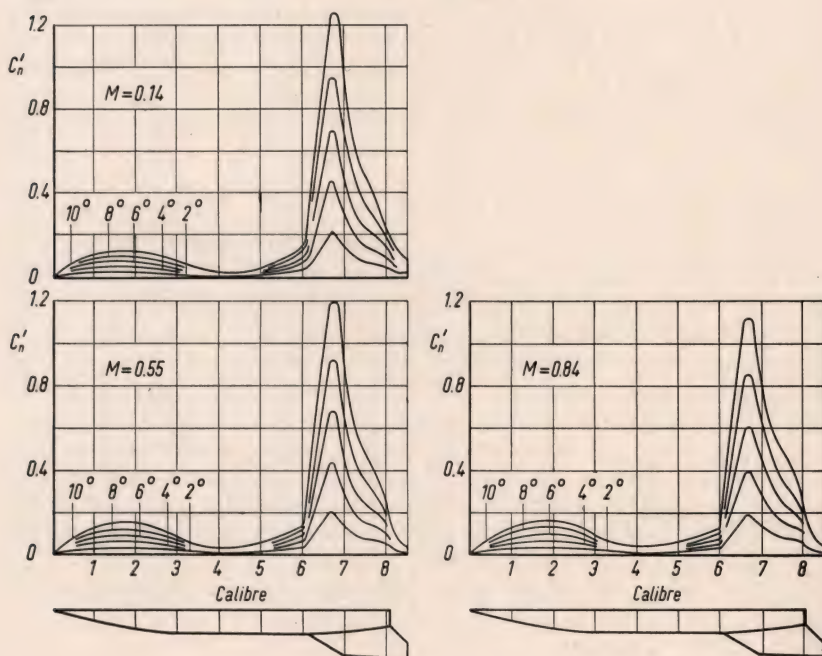


Fig. 9. Normal force distribution for A-4 in subsonic flow

These curves revealed the interesting fact that the pressure distribution over the fins is quite different in subsonic and supersonic flow. The supersonic fin shows a more uniformly distributed normal force over the whole area whereas in subsonic flow, as found before, the force is concentrated close to the shoulder. It can also be seen that the pressure on the fins decreases faster than the pressure over the front part of the body with decreasing angle of attack. This demonstrates the non-linear restoring moments, or the shift of centre of pressure

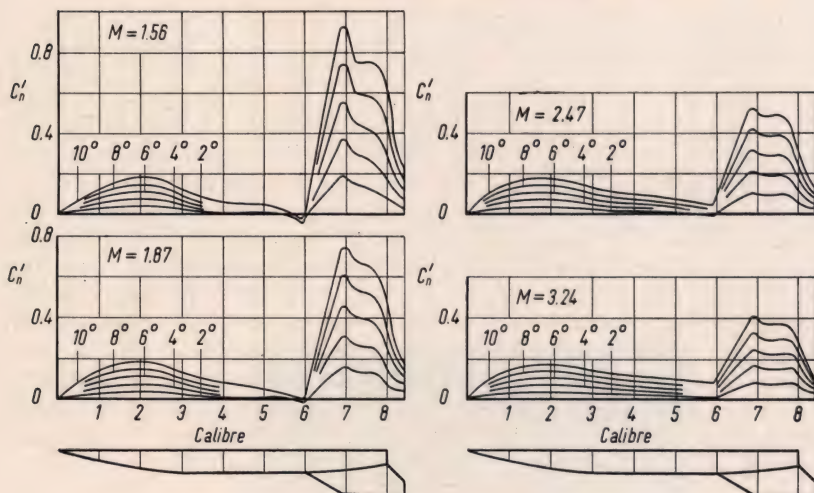


Fig. 10. Normal force distribution for A-4 in supersonic flow

with angles of attack, which had been discovered in the previous force measurements.

#### 2.1.7. Possible Instability Regions of the A-4 Rocket

It was somewhat alarming that, with increasing MACH number, the centre of pressure gradually approached the centre of gravity of the rocket. The measurements at  $M = 3.3$  showed that the rocket should still be stable, but the trend of the curve indicated increasingly less stability up to  $M = 5.2$  — the calculated maximum MACH number for the full-scale A-4. The test point at  $M = 4.4$ , which was obtained later, was not more encouraging. In other words, the pitching moment curves changed slowly from type *a* to type *b*, see Fig. 11,

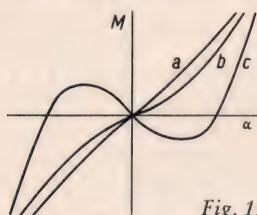


Fig. 11.

and the only hope was that it would not turn from the point stable case into type *c*, the circular stable condition. A similar uncertainty prevailed in the transonic region, where no measurements at all could be made between  $M = 0.9$  and  $M = 1.2$ , and where slight instability was also indicated at low angles of attack (Fig. 6).

For some time the fear of this transonic instability was supported by observation of the small test rockets, which could not exceed the speed of sound because of the very pronounced “corkscrew” motion described above.

There was some scepticism among the aerodynamicists when the guidance control people indicated that slight instability of the roll-controlled rocket could be tolerated by the control equipment. We all crossed our fingers during the first start of the full-scale A-4, hoping that the short time to pass the transonic region would not allow a dangerous yawing.

## 2.1.8. Jet Effects on the Aerodynamic Characteristics

In addition to the above damping measurements, some investigations were made about the influence of the jet on the drag of the A-4. Pressure distribution models with compressed air jets were constructed, and measurements at simulated pressure ratios (model tank pressure to test section pressure) were carried out. In a very simplified manner, in which only geometric similarity of the jet expansion at corresponding altitude and MACH numbers could be obtained interesting information on the jet influence on drag became available.

The first calculations on the expansion of the rotational symmetric jet, using the method of characteristics, indicated an alarming expansion angle of approximately  $70^\circ$  at the "cut-off" altitude, approximately 25 km, much larger than the  $45^\circ$  sweep of the trailing edge could stand without being burnt. However, in the great hurry, the calculations had been made without external flow, and according to subsequent calculations the still noticeable dynamic pressure at the "cut-off" points actually squeezed the expansion angle below the  $45^\circ$  trailing edge angle. An experimental check of this result was very desirable. A model, with only a few pressure holes along a meridian, was mounted on a profiled side strut through which not only the pressure tubes, but also the compressed air was led in. The exit nozzle was similar to the full-scale nozzle with an expansion ratio of approximately  $M = 2.0$ . The influence of the side strut, though not negligible, was apparently small compared with the effects of the jet itself. Pressure measurements at zero angle were made alternately with jet on and jet off and the difference evaluated as a drag difference. The jet configuration, and therefore the drag difference, varied considerably with the wind tunnel MACH number and the internal pressure. In Fig. 12, curves calculated from the wind-tunnel results for the conditions of the 300 km trajectory show the drag difference. It can be seen that throughout

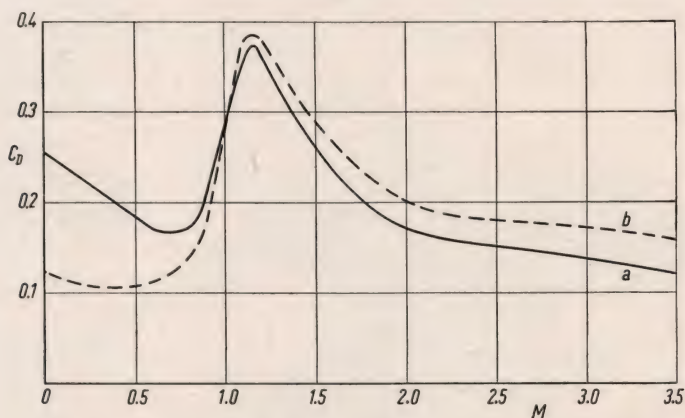


Fig. 12. Drag coefficients of A-4 with jets (a) and without jets (b)

the subsonic region the drag increases considerably, whereas in the supersonic region the drag is reduced by the jet. In subsonic flow, at relatively low altitude with little jet expansion, the jet accelerates the ambient air over the ogival afterbody, thus lowering the pressure and increasing the drag. In supersonic

flow at great heights the jet is expanded and, therefore, the ambient air passes through a compression shock at the base of the body. It can be seen in Fig. 13 that, due to the pressure increase, the boundary layer becomes detached, the ogival tail pressure is raised, and thus the drag is reduced.



*Fig. 13 a. Expansion of jet at high altitude ( $M = 0$ )*



*Fig. 13 b. Expansion of jet at high altitude ( $M = 1.87$ )*



Fig. 13 c. Expansion of jet at high altitude ( $M = 3.24$ )

The photographs show that the models had a laminar boundary layer. No corresponding tests were made with a turbulent boundary layer, which might have changed the flow configuration considerably.

#### 2.1.9. Special Tests on Enlarged Parts of Models

The 40-cm tunnel was a well-chosen size of tunnel for force, pressure, and dynamic measurements of our rocket models. For a few investigations, however, it was too small. For example, the hinge moment measurements of the jet vanes could not be made with reasonable accuracy using the full-size jet model, since in the model scale the rudders measured only a fraction of a square centimetre. To overcome this difficulty only the rear part of the model was used with its diameter increased three times. It was mounted on a tube reaching from the subsonic side of the nozzle, through the nozzle throat, into the test rhombus. Compressed air was fed through the tube into the model. The rudders were now large enough for the moments to be measured easily, and they were determined, with a special hinge moment balance, for several sets of vanes which represented consecutive stages of flame effects on the burned shoulders.

This trick of measuring small components on enlarged body parts was applied later to the model testing of other guided missiles, "WASSERFALL", "RHEINTOCHTER", etc. with good success.

## 2.2. Thermodynamic Data

### 2.2.1. Basic Problems

Originally, the rocket skin material was selected solely for mechanical reasons in order to keep the weight down as much as possible. However, the calculations

which G. EBER carried out in the early days at Peenemünde<sup>13</sup>, indicated clearly that the skin metal had to withstand unexpectedly high temperatures. Compression and friction heat of the air was estimated to produce a temperature of more than 1,000° C at  $M = 5.0$ . How much of this heat would enter the skin was, however, a more or less open question. Very few theoretical results were available and no experimental data had been previously obtained at these supersonic MACH numbers.

### 2.2.2. Wind-Tunnel Tests

In order to calculate the skin temperatures exactly, the two basic factors, the heat-transfer coefficient and the potential temperature difference, in the heat-transfer equation:

$$Q = h (T_e - T_w) \quad [\text{cal/m}^2 \text{ h}],$$

where  $Q$  is the amount of heat per unit time and area, had to be determined. Since the wall temperature  $T_w$  increased continuously during flight, the heat-transfer rate depended on  $h$  and the effective boundary-layer temperature  $T_e$ . An experimental program was, therefore, set up to measure these quantities on cones, cylinders, and flat plates in the supersonic tunnel, since these shapes were more easily adapted to theoretical calculations than the rocket models. Before the Peenemünde tunnel was in operation, A. NAUMANN<sup>14</sup> ran tests on small cones in the Aachen  $10 \times 10$  cm supersonic tunnel. These measurements were afterwards repeated at Peenemünde by G. EBER<sup>15</sup> on larger cones and over a wider range of MACH numbers; the dimensionless recovery factors were plotted as

$$r = (T_e - T)/(T_0 - T)$$

as a function of MACH number, where  $T$  is the free stream, and  $T_0$  the adiabatic stagnation temperature.

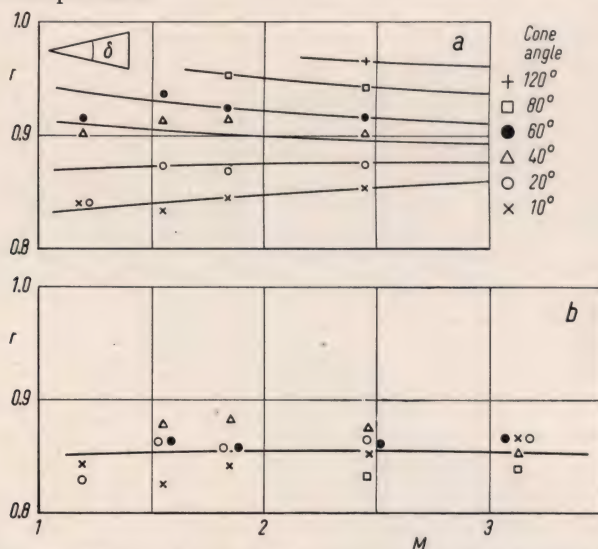


Fig. 14. Recovery factors of cones with various cone angle at laminar boundary layer  
a) Free-stream conditions    b) Condition on cone surface outside of boundary layer

EBER obtained the interesting result, shown in Fig. 14, that the recovery factor, when evaluated from free-stream conditions, varies considerably with cone angle, and that the different values can be brought together when flow conditions behind the cone shock were used.

It can be seen that these recovery factors were close to

$$\sqrt{Pr} = 0.84$$

where

$Pr = c_p \mu / k$  is the PRANDTL number,

$c_p$  is the specific heat,

$\mu$  is the coefficient of viscosity and

$k$  is the conductivity.

This value of  $\sqrt{Pr}$  agrees with the theoretical and experimental values for a laminar boundary layer, but the values for the turbulent boundary layer turned out to be somewhat higher than the now accepted figure of  $Pr^{1/3}$ .

The measurements of the heat-transfer coefficients were more difficult, owing to the restricted running time of  $18 \times 20$  sec. However, heat-transfer coefficients in the form of NUSSELT number could be satisfactorily obtained from the slope of the transient curves,  $T = f(t)$ , the heat capacity, and the mass of the copper models. The data points were scattered with reasonable accuracy about the straight line (in logarithmic presentation):  $Nu = 0.0107 Re^{0.82}$ . They turned out to be independent of MACH number and a function only of the PECLET number,  $Pe = Re \times Pr$ , if the properties of the air at surface conditions were taken in the determination of REYNOLDS and PRANDTL numbers. Therefore, the extrapolation of the wind-tunnel results to the full-scale MACH 4 and 5 conditions seemed to be justified, and, using these values, skin temperatures of the full-scale A-4 were computed for several metals such as dural, steel, monel, etc. of various thickness. The example shown in Fig. 15 was calculated on the assumption of a turbulent boundary layer, taking into account the estimated effects of radiation, in addition to the convection heating. Later full-scale tests

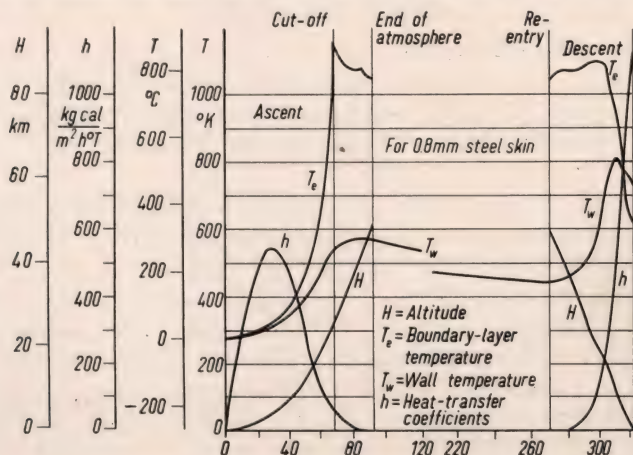


Fig. 15. Calculation of wall temperature

showed that the calculations were not too far from the skin temperatures actually obtained.

### 3. CONCLUSION

During the last decade, considerable progress has been made in aerodynamic science and technique, and this makes the original work appear to be simple and the aerodynamic and thermodynamic results inadequate for modern rocket design. But this pioneer work helped the V-2 to fly, and it prepared the ground for the future investigations in which we are still involved. It stimulated hundreds of later investigators to continue this work, to repeat the measurements, and to find new knowledge for a technical development, in which the V-2 rocket was only step number one, — and which was well done!

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## DEVELOPMENT AND TESTING OF THE V-1 AUTO-PILOT

HEINRICH TEMME \*

Probably the best way of introducing this paper is to give an outline of the problems which led to the development of the V-1.

The problem was roughly as follows: to develop a small, unmanned, expendable aircraft using an internal-combustion engine of a certain design. The aircraft was to have an explosive head, and crash after a flight of up to 250 km on to a target decided in advance. It was to be possible to launch the missile from a catapult or from a carrier aircraft and it was to be able to fly at altitudes between 300 and 2500 m and to reach objectives situated  $\pm 60^\circ$  off a straight course.

The accuracy demanded meant that the 50% longitudinal and lateral divergence should be not more than 4% of the distance flown, i. e.  $\pm 4.5$  km at 225 km range. This accuracy had to be achieved only by the auto-pilot; additional radio-control was not to be used because of possible enemy interference.

This and the characteristics of the airframe led to some special requirements which the auto-pilot had to fulfil; these will now be discussed. The auto-pilot had:

1. to improve lateral stability,
2. to control the heading for about 25 min.,
3. to allow direction changes of  $\pm 60^\circ$  after launching, when shooting laterally,
4. to maintain a fixed altitude between 300 and 2500 m, and finally
5. to get the V-1 down after having travelled a distance decided in advance.

For this last mentioned problem a special air mileage measuring unit, an air-log with electrical counter device, was used. This device was set manually for the predetermined distance to the target. When the given distance was reached in flight the missile elevators were deflected to cause the missile to dive.

The operating conditions of the V-1 called for certain additional characteristics, e. g. insensitivity to the launching acceleration of 22 g maximum and to vibrations introduced by the engine. Inspection and adjustment immediately before take-off should be easy and not take too much time. Since, like all other components of the V-1, it was going to be lost, production should be easy and cheap.

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At the project stage of the V-1, no suitable auto-pilot able to fulfil the foregoing conditions existed. However, the ASKANIA Co. had started the development of a simple auto-pilot for missiles with pneumatic auxiliary power. This development, which had been originated shortly before by the Air-Ministry expert Mr. EVERS, was taken over. The choice of a pneumatic auto-pilot was also favoured by the fact that the question of a power supply was relatively simple, for compressed air had already been provided for the fuel system.

This auto-pilot was a rudder and elevator control only for directly correcting any heading and altitude disturbance and for indirectly correcting a roll disturbance. There was no special roll control system. During the project stage there was the opinion that by careful production and adjustment of the wings the disturbing moments about the roll axis could be kept small compared with the moments about the yaw axis, which were caused mainly by an asymmetrical fitting of the power unit. Considering the small lateral divergence required it was hoped also to be able to abandon the aileron actuation by the heading control device or by an additional roll control system.

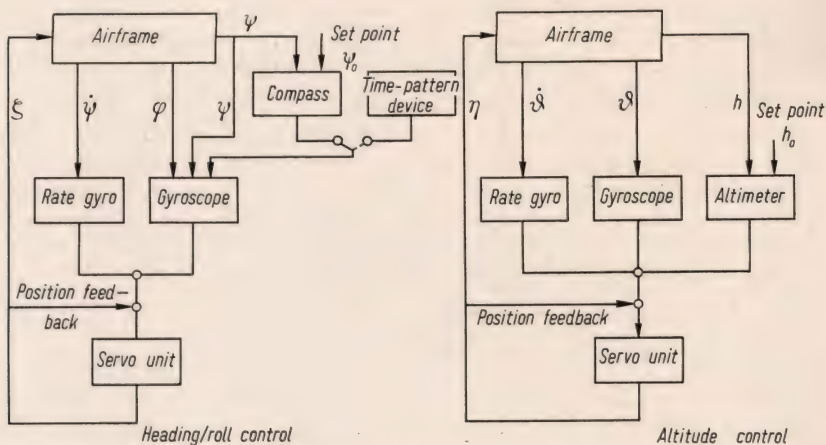


Fig. 1. Schematic diagram of heading/roll and altitude control

Fig. 1 is a block diagram of the controls and the missile itself. Each of these is what is known as a control loop. The connecting lines between missile and regulator exit and entrance do represent this loop. The diagram on the left gives the control loop for heading/roll, the diagram on the right for altitude.

Three terms were used for the heading/roll control: heading or azimuth, rate of angular movement in the yaw plane, and bank angle. Azimuth and bank references were provided by a compass-monitored gyroscope. The signals from these references as well as from the rate of yaw reference were added in a certain proportion and fed into a servo-unit with position feedback. This unit moved the rudder. A time-pattern device gave the signal for turn. A bank signal caused the control to induce a rudder movement which generated a skidding action. This produced a roll moment which tended to bring the missile back into a level attitude. This sensitivity of the control system to bank improved lateral stability, as was shown by calculation.

The automatic heading and roll control using one channel was not new. Mr. MEREDITH mentioned an arrangement like this much earlier. During the first years of the war and before, different research institutes in Germany examined this arrangement in the course of their theoretical investigations into the control of manned and unmanned aircraft.

The right part of Fig. 1 shows the altitude control. No special precision was required of this control. A deviation of  $\pm 100$  m was not considered serious, as long as it did not appear as a periodic oscillation.

This control also employed three terms: altitude, which is the controlled variable, pitch, and rate of angular movement about the lateral axis. The values of the three terms were summed and again fed into a servo-unit, which in this case adjusted the elevator.

As is well known, two sorts of oscillation are caused by a disturbance of the longitudinal motion of an aeroplane: a slightly damped oscillation of the flight path, which has a long period and is called phugoid motion, and a heavily damped oscillation of the angle of attack with short period.

Had this control measured only altitude, then the amplitude and period of the flight path oscillation would have become smaller, but the damping would have become still more unfavourable than without control because of the merely proportional effect of the control. It was the derivative action of both the additional measuring devices which made the oscillation well damped.

In automatic control technology it is customary to gain the derivative action either by measuring the rate of change of the controlled variable, or by differentiating it, or by a lag in the feedback path. This would bring about difficulties in measuring technique and design in regard to the altitude control. Therefore in this case one gains the derivative action by an auxiliary-controlled variable, which corresponds to the rate of change of altitude.

This auxiliary-controlled variable is the pitch. It is easily and exactly measured by a gyroscope. Unfortunately, this introduction of pitch has a disadvantage. The damping of the fast oscillation of angle of attack, the frequency of which is increased further by the influence of the auxiliary-controlled variable, is reduced strongly by the lag in the control mechanism. In order to compensate for this unfavourable effect, the rate of angular movement about the lateral axis was added.

Fig. 2 is a schematic diagram of this auto-pilot. Some of its components have been omitted in order to get a better general view.

In the upper left hand corner is seen a displacement gyroscope, which measures all the required angular values. Its outer gimbal ring is mounted in bearings parallel to the lateral axis, and during climb is inclined at  $15^\circ$  to the transverse plane of the missile. The inclination increased by  $5$  to  $6^\circ$  during changing into the horizontal flight. The movement of this outer gimbal ring measured relative to the case will yield the pitch which is then fed as an auxiliary-controlled variable into the altitude control device. Owing to the inclination of the outer ring the axis of the inner ring is not parallel to the vertical axis. Thus its angular movement relative to the outer frame indicates not only the heading but a bank component too. The ratio of bank component to directional component will be determined by the inclination. The data observed by the gyro were detected pneumatically so that they could be added

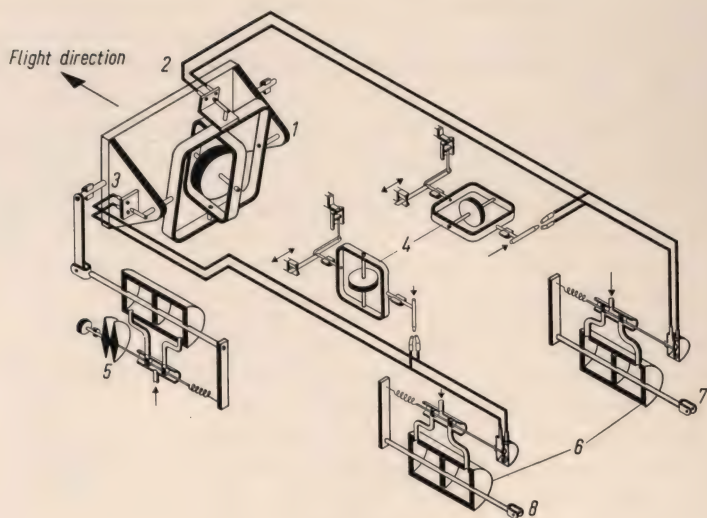


Fig. 2. Principle of the V-1 auto-pilot (Askania Co.)

- |                             |                              |
|-----------------------------|------------------------------|
| 1 = Displacement gyroscope  | 2 = Pick-up for heading/roll |
| 3 = Pick-up for pitch       | 4 = Rate gyros               |
| 5 = Altimeter with actuator | 6 = Servo units              |
| 7 = To rudder               | 8 = To elevator              |

to other measured quantities. The position of the inner gimbal ring was monitored by a remote magnetic compass with pneumatic transfer, i. e. the compass made the gyro precess slowly at a rate of  $3^\circ$  per minute if the direction of the missile did not coincide with the compass set point. The precession then made the directional control turn the missile, until coincidence was obtained. (This compass had already been successfully employed in manned aeroplanes. The sort of difficulties which appeared when using it in the V-1 are discussed below.) The differential pressure picked up at the compass controlled a two-way electrical switch by means of a diaphragm. This switch, according to its position, turned on one of two torque-creating circuits not shown in Fig. 2. These torques, acting upon the outer gimbal axis, made the gyro precess about the inner axis.

The rates of angular movement about the normal and the lateral axis are detected by two spring-restrained gyros, with pneumatic pick-up. A flat spring, having an adjustable free length, exerts this restraint and so it was possible to vary the effect of rate of angular movement upon the control process. Another means of accommodating the control devices to the missile requirements was by altering the pneumatic pressure of the pick-up.

The pneumatic pick-ups of each channel are fed in parallel into a differential pressure element. The differential pressure amounted to about  $\pm 100$  mm of water. The diaphragm in the element acts upon a control piston which in turn controls the servo power, i. e. compressed air of 6 atm, to a servo-unit. This operates the control surface and is connected to the control piston by a restoring spring. The time required for the servo-unit to travel between its end positions

is about 1 to 2 sec, its maximum force being 70 kg. With the standard version the control piston was arranged vertically to the direction of flight, because the trials had shown that the acceleration at the start produced control signals not intended.

Directional changes which were to be brought about when shooting laterally took place during climb at  $1^\circ$  per second by means of gyro precession. The directional control mechanism had a time pattern device for this purpose, which is not shown. This device fed into the torque generator at the outer gimbal ring instead of the compass, and made the gyro precess about the axis of the inner gimbal ring, which is inclined to the perpendicular. This inclination makes the pitch decrease with directional changes, which effect however is gradually cancelled by the pendulous unit monitoring. Since, as a consequence of this procedure, the rate of climb decreases, turning begins after a flight of 1 to 2 sec only. The turn being finished, the monitoring action of the compass cuts in and, if necessary, corrects the angle of precession.

During climb the pitch was kept at a constant value of  $7.5^\circ$  by a signal from the displacement gyroscope. The deflection into the level flight was initiated at about 275 m below the required altitude. This deflection was achieved by the altitude control system with aneroid elements. The latter control a pneumatic actuator, which is also employed as a servo-unit, having position feedback. The actuator displacement is proportional to the altimeter reading. This actuator revolves the case of the displacement gyroscope, which means a change of

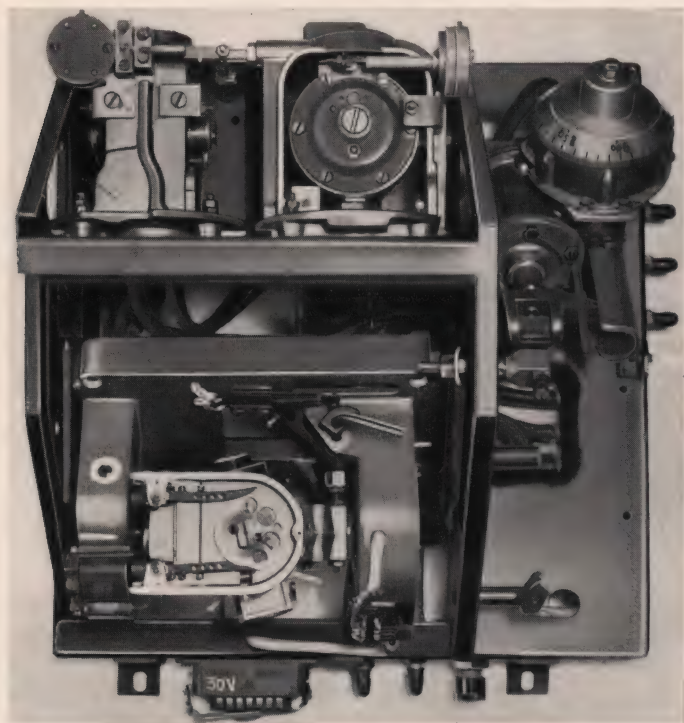


Fig. 3. View of "Steuergerät" (control device)

position of the pick-up for the pitch, and this again causes the aneroid system to call for a change of altitude.

The displacement gyroscope, rate gyros and altitude measuring device and its actuator were designed as a single unit called "Steuergerät" (Fig. 3).

Fig. 4 shows how the auto-pilot was built into the air-frame of the V-1. At the front is seen the compass mounted within a wooden sphere, the significance of which will be explained below. Behind the front mounting for the engine is the "Steuergerät" easily accessible from outside. In the rear are seen both of the servo-units.

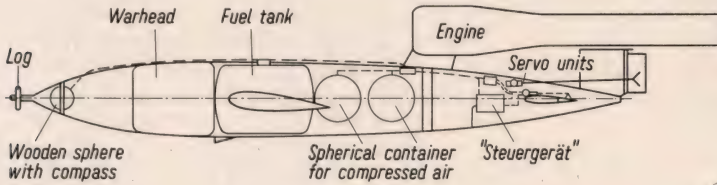


Fig. 4. V-1 with auto-pilot

The compressed air supply was taken from two spherical bottles, which also had to furnish the air for the fuel supply. These bottles were filled up to 150 atm before take-off. The air pressure was reduced to 6 atm at the bottle exit and within the "Steuergerät" a further reduction took place for the gyros to 1.2 atm. The air consumption was 300 litres per minute.

The electrical power needed for the monitoring of the gyroscope and for the time pattern device was quite small.

The general laboratory testing of the auto-pilot was carried out by the ASKANIA CO. There it was possible to investigate also its behaviour with 25 g accelerations. Further laboratory work was done at the testing facilities of the LUFTWAFFE at Peenemünde and with the DEUTSCHE FORSCHUNGSANSTALT FÜR SEGELFLUG at Ainring.

The calculation of stability was carried out by the DEUTSCHE FORSCHUNGSANSTALT FÜR SEGELFLUG at Ainring.

It is well known that the flight testing of the auto-pilot was done at the flight testing centre of the LUFTWAFFE at Peenemünde. Unfortunately this testing had to be done simultaneously with the testing of the airframe, engine and catapult. For this reason troubles were very often difficult to isolate: some were never cleared up. It was also a disadvantage that the experiments were made over the sea, so the missile could not be retrieved and examined to find the reasons for failures.

Most of the V-1 missiles were launched by catapult, a few only from a carrier-plane. At first, when short distances up to 30 km were sufficient for testing, the flights were carried out in a northerly direction. Later, after the distances extended up to 250 km, the flights went along the coast of Pommern.

Altogether about 300 test-flights were made before operational employment. In these tests the character of the flight-path, the attitude of the missile and the positions of its rudder and elevator were taken as indications of the quality

of the controlling process and of the behaviour of single control components by different methods.

A film was taken of each take-off. From this it was possible to study the flight path and attitude of the V-1 during the first stage of the flight. In the second stage, still in the range of optical sight, the path was checked by cine-theodolite and out of the optical sight by radar. Occasionally the last part of the flight path was also checked by a small, long-wave transmitter of which a bearing could be taken from the ground. A few of the test missiles — about 40 — were fitted with a one-channel, 4 Mc/sec transmitter with which different data, e.g. attitude, altitude and position of the controls could be transmitted to the ground stations or to an accompanying aeroplane. For some data additional measuring equipment had to be built in. All test values were picked up electrically by means of a resistance type detector and were fed into the transmitter through a switch-gear. This switch-gear swept through all the values 10 times per second. The signals were recorded in the receiver station by an oscillograph.

The test procedure was as follows: at first the behaviour for the V-1 was studied during short climbing only, the auto-pilot operating without compass monitoring, without aneroid elements, and without time-pattern device for turning. Then the altimeter was added so that investigations could be made about the transient from climb into level flight, the altitude control, and the function at higher speeds. Later, compass monitoring was included; by this means it was possible to estimate the achieved lateral divergence. Finally, for testing turn, the time-pattern device was installed.

It was an essential part of the testing to find and also to remove the unfavourable effects of the high accelerations at take-off and the heavy vibrations of the engine upon the airframe and auto-pilot.

Some of the problems which arose during the tests will now be described.

The first short climbing flights of the V-1 demonstrated that longitudinal and lateral stability were sufficient. Auto-pilots had been built in the axis of the inner gimbal ring which, at the launcher, was parallel to the normal axis of the missile. Since it did not yet have an inclination like the one in the missile shown in Fig. 2, the bank could not be measured. Calculations had shown that stability, even though small, was to be expected. After equipping the missiles for the next flights with the pendulous unit hitherto not built in, and directing the spin-axis horizontally, the missiles crashed because of lack of lateral stability. This phenomenon was particularly pronounced when taking-off with a cross-wind. The horizontal position of the spin-axis meant that now during climb the axis of the inner gimbal ring was inclined in the reverse sense towards the normal axis of the missile as shown in Fig. 2, so that an angle of bank caused a controlling signal to be given, which instead of righting the missile increased its bank until it crashed. The calculation of stability enabled the failure to be quickly recognized and avoided.

It took quite a while to test the compass. A big problem was the elimination of the strong aberration originating from the airframe, which was built almost entirely of sheet steel. It was impossible to allow for this when adjusting the compass, for the engine vibrations during flight made it decrease indefinitely. In order to get only a small but known deviation, those parts of the airframe close to the compass were knocked with wooden hammers, after the airframe

had been adjusted to the desired course within a room which contained no iron. The momentary matrix-loosening effected by the knocking allowed most of the magnetic dipoles to take up the direction of the earth's magnetic field. In this way the aberration was decreased to  $1^\circ$  and less.

A second problem, when testing the compass, was how to eliminate errors of up to  $\pm 10^\circ$  in the readings which appeared when the engine was running. It was found that mechanical vibrations and sound waves from the engine were the cause. It was possible to diminish this error to between  $0.5$  and  $1^\circ$  by isolating the compass from the vibrations and reducing the friction of the magnetic needle. The isolation had to be effective against both sorts of vibration mentioned above, so the compass was suspended from rubber springs within a wooden sphere, the sphere itself being suspended from the airframe by helical springs. The springs absorbed the mechanical vibrations; the wooden ball did the same with the sound vibrations.

Great difficulties arose, for instance, in recognizing the cause of crashes which frequently occurred after the transient of the missile from climb to level flight. It was not the auto-pilot that was responsible for these crashes, as it had first been assumed, but a rolling moment of the airframe, caused by a difference between the angles of attack of both wings due to the acceleration at the take-off. The auto-pilot was not able to compensate this rolling moment.

After eliminating the cause for the above failure still quite a number of missiles were flying with a bank of  $2$  to  $7^\circ$  caused by misalignment of the wings. The bank angle produced considerable skidding and a corresponding lateral divergence. Since due to lack of time, larger alterations, with the aim of reducing the bank angle, could not be carried out on the airframe and the auto-pilot, the FIESELER CO. proposed the additional application of an integral bank control, which made use of a pendulum and the dynamic pressure as auxiliary power. This pendulum controlled the pressure, which was then fed into a small turbine operating a trim flap for banking through a gear reduction.

A lateral 50% divergence of  $\pm 4.5$  km ( $1.2^\circ$ ) at 225 km distance had been requested at the beginning of the tests. After the preliminary testing, shortly before going into service, the lateral divergence was about twice that value. The results were said to have been slightly better when taking-off from a carrier-plane.

There were two reasons mainly for the large lateral divergence: the disturbing moments of the airframe about the roll axis, and the compass errors, the first one being predominant. There was no time to approach these problems systematically by calculation and tests.

The costs required for the production of the auto-pilot were large at first, especially as most of the components were designed as light alloy castings. Since considerable numbers were needed, the Air Ministry demanded that the design be changed in order to cut the production cost. This change brought about:

1. an essential simplification of some components e. g. the pick-up mechanism on the displacement gyroscope,
2. a replacement of the light-alloy castings by sheet steel design.

The production costs were considerably reduced by these measures. As far as the author remembers not all of the components of the auto-pilot came into action in this simplified form.

The work necessary for this preparing for mass production absorbed most of the ASKANIA Co.'s development capacity. Consequently, further development could only be carried out to a limited degree. It aimed at a new compass and a rate gyro with an integrating device.

The new compass was to be less sensitive against engine vibrations. Principle tests were made with a compressed air bearing for the needle system. Within a half-spherical shell of 2 to 3 cm radius another similar shell was floating, supported by compressed air and carrying the needle system. The friction between the two shells was extremely small. It was hoped that no special packaging for shipment and no sound isolation in the airframe would be necessary for this small and sturdy compass. These tests had not been concluded by the end of the war. Only a few specimens had been flight tested.

Careful production methods are needed for the satisfactory functioning of the displacement gyroscope. That is why experts at different places in Germany, particularly with LUFTFAHRTGERÄTEWERK HAKENFELDE, considered the question of how this gyroscope might be replaced. It was decided that the attitude should be detected by integrating the rate of angular movement. In this way the rate gyro with integrating device originated.

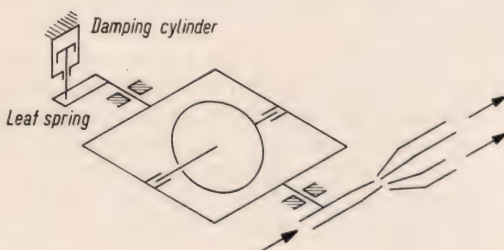


Fig. 5. Rate gyro with integrating device

Fig. 5 shows a design for such a gyro proposed by the ASKANIA Co. The rate gyro again has a flat spring, which is, however, no longer fixed to the case but to a piston moving within a cylinder filled with oil. The piston velocity is proportional to the force exerted by the gyro through the flat spring, and this force is again proportional to the rate. Consequently, the piston velocity and the rate of angular movement are proportional to each other. Thus the position of the piston equals the time integral of the rate of angular movement, i. e. it is equivalent to the angular movement. The gyro deflection is in this way proportional to the angular movement and to the rate of angular movement. The piston velocity depended very much on the temperature, because there was no silicone oil. In order to avoid the disadvantage of the temperature dependency of the oil damping, tests were made too with eddy-current damping.

The displacement gyro and both of the rate gyros were intended to be replaced by two of the integrating gyros just described. Need of space and of power in the airframe, as well as the production costs, would have been essentially less for these gyros than for the former arrangement.

All these investigations of the new compass and of the rate gyro with integrating device got no further than the preliminary stage at the ASKANIA Co.

## DISCUSSION

Mr. A. R. WEYL (Dunstable, Beds): Speaking from the receiving end, I should like to ask if the behaviour of the V-1 over the target has indeed been as intended by the dispatching side. We experienced that over the target the elevator was depressed for making the aircraft dive into the ground. At the same moment when the elevator was depressed, the chugging noise of the propulsion ceased abruptly, and this provided a very welcome warning to take cover. At the time, we did not believe that it had been intended to convey such warning signal; it seemed to come about by starving the pulse duct when the dive was induced, on account of the remaining fuel being thrown up in the tank. Unfortunately, the then Home Secretary saw fit to broad-cast over the radio that the V-1 had the peculiarity of giving a warning before descending upon "Southern England". It would have been wiser to communicate his advice more discreetly through the Police and the Civil Defence organization. As it was, subsequent V-1 missiles were observed to sneak in with long, silent approaches, whilst others spiralled in with the pulse duct going full blast.

It would be interesting to learn how much of this behaviour was intentional.

Dipl.-Ing. H. TEMME: This behaviour of the V-1 after the beginning of the dive was not intended by us. We found the cause to be the same as Mr. WEYL suggested. Since we did not at first suppose that the behaviour mentioned above could be used as a warning signal we did not take steps to remedy it until later.

# THE CONTROL SYSTEM OF THE V-2

OTTO MÜLLER \*

## 1. THE CONTROL PROBLEM AND THE REQUIRED CONTROL CIRCUITS

The long range rocket V-2 was designed to transport its warhead to a target about 250 km away. It was to reach this target as an unpropelled projectile after the thrust had been cut off. Simple calculations showed that the maximum range for a given velocity at all-burnt was obtained with an angle of  $49^\circ$  between trajectory tangent and the vertical. Because of the starting weight of 12.5 tons, the normal type of inclined launching ramp would have been too heavy. Thus the rocket had to undergo a change of direction after launching; for this a control loop, the so-called pitch control, was required.

Furthermore, a control in yaw had to be incorporated, for without this the rocket would have deviated too much from its intended trajectory in the lateral direction.

A third control loop was required to suppress the natural tendency of the rocket to roll due to structural inaccuracies. It was not permissible to let the V-2 roll, because, as we shall see later, the control equipment contained gyros.

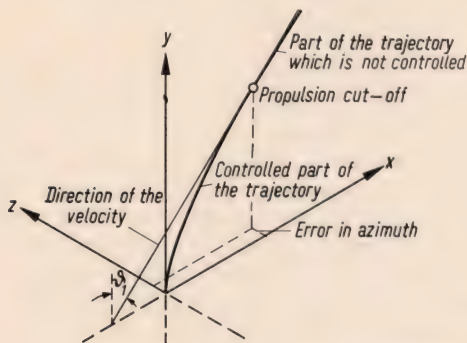


Fig. 1. The first part of the trajectory of the V-2

In order to obtain the desired range, the thrust — after the end of the directional change — had to be cut off when the rocket had reached a predetermined velocity. For this purpose, a device had to be developed by which the velocity of the rocket could be measured and the fuel shut down,

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when the "all-burnt-velocity" had been reached. Fig. 1 shows the first part of a trajectory of a V-2 rocket.

## 2. THE CONTROL EQUIPMENT FOR THE STANDARD SERIES

### 2.1. The Automatic Pilot for Yaw, Pitch and Roll

For the directional control of the rocket, the automatic pilot with gyros as used in aircraft readily suggested itself. We shall consider the control in yaw first. A potentiometer attached to a gyro produced a voltage proportional to the angle between the actual and intended direction of the axis. The voltage, via a stabilizing network and two amplifiers in parallel, operated a couple of servo units (Fig. 2).

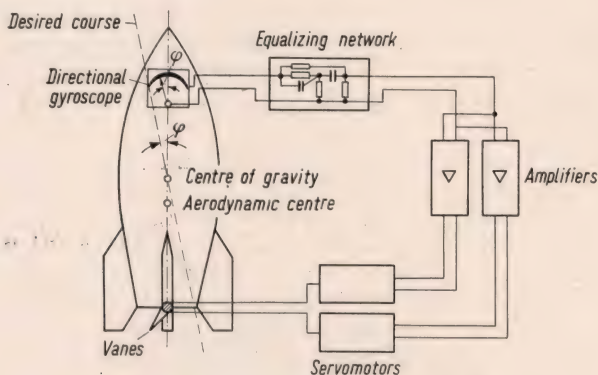


Fig. 2. Scheme of the control in yaw

The particular difficulties in designing the control circuit were as follows:

- Only a velocity servo was available as a servo drive, since at that time it was used in aircraft.
- The aerodynamic forces varied considerably during flight. The velocity of the rocket and the density of the air varied by several orders of magnitude, and the distance between the centre of pressure and the centre of gravity by a factor of 10.

However, a solution was found, in which a passive network was used and the amplification as well as the transfer function of amplifier and network were kept constant from start to all-burnt, this being extremely desirable for technical reasons. In order to avoid tedious repetition we shall consider at a later stage how this solution was found.

The degree of feed-back was chosen so high that the control circuit was just sufficiently damped throughout the control time. This is not *a priori* the best solution: an accelerated unguided rocket veers into the wind, whereas a guided rocket with a high degree of feed-back in its controls drifts with the velocity component of the wind normal to the assigned trajectory. Therefore there is a certain degree of feed-back for which the wind has no influence on the lateral motion. Calculations, however, showed that for all disturbing forces occurring

laterally, the accuracy in yaw was best in the case of highest possible degree of feed-back in the control circuit.

Graphite controllers were used as control vanes in the venturi with, for reasons to be explained later, small air-vanes in parallel. Controllers in the venturi were used for two reasons:

- a) The relatively long time during which the rocket had to travel, because of its small initial acceleration, in order to develop at the air-vanes reaction forces sufficient to counterbalance wind and thrust unsymmetries, would have been sufficient to make the rocket turn through  $90^\circ$ .
- b) The available hydraulic servo units were of rather low performance. It was therefore necessary to use well balanced controllers, i. e. of a type with a very small torque. It was not thought possible that within permissible time air-controllers could have been developed which, for the whole velocity range (MACH number 0 to 6), would be sufficiently balanced.

After many trials, graphite controllers of such a shape were developed for use in the venturi that, in spite of loss through burning, equilibrium was sufficiently retained for the whole of the trajectory. One may say that the low performance of the servo units prevented the development of a really good yaw control circuit for the V-2.

The control in pitch was nearly identical with the control in yaw. Only the direction assigned to the roll axis (direction of course) was varied in time so that the desired direction of the trajectory and thus the desired change in direction was obtained. The direction of the course and the trajectory tangent were nearly identical in the case of the V-2-control systems. For the chosen trajectory the angle of attack remained always smaller than  $3^\circ$ , and the angle between the axis of the rocket and its assigned direction never exceeded the same value because of the high degree of feed-back in the control circuit. The accurate calculation of the assigned direction from the desired trajectory tangent is not difficult.

In the first controlled rockets the direction of the course was changed by making the gyro precess by a signal (the precession axis was called *D*-axis). The direction of the course was the direction of the spin axis of the gyro. Later on, the pick-off pitch was given as angular deviation with respect to the rocket; in this method, which yields better results, the direction of the course differs from the axis of the gyro by this angular deviation.

The control circuit for roll was identical with the control in yaw up to the amplifier input, only the network was somewhat different. The output voltage of the network was fed into both amplifiers for yaw control, from which one feed line passed through a phase-inverter. The roll signal thus operated the yaw controllers in phase opposition. We now understand the purpose of the small additional air controllers: because of their small leverage about the longitudinal axis the effectiveness of the controllers in the venturi was not sufficient to counterbalance the moments in roll. It was found that even the additional air controllers had insufficient effect. Therefore a so-called "Trimmsegelsteuerung" was built into the tail plane which, by way of a control circuit making use of integrated deviation, compensated continuous disturbances in roll. Fig. 3 shows a circuit diagram of the direction and roll control circuit. We see that a gyro, a so-called "Vertikant", carried the yaw and roll pick-offs; the amplifiers and networks were combined in the so-called mixer unit (summing amplifier)

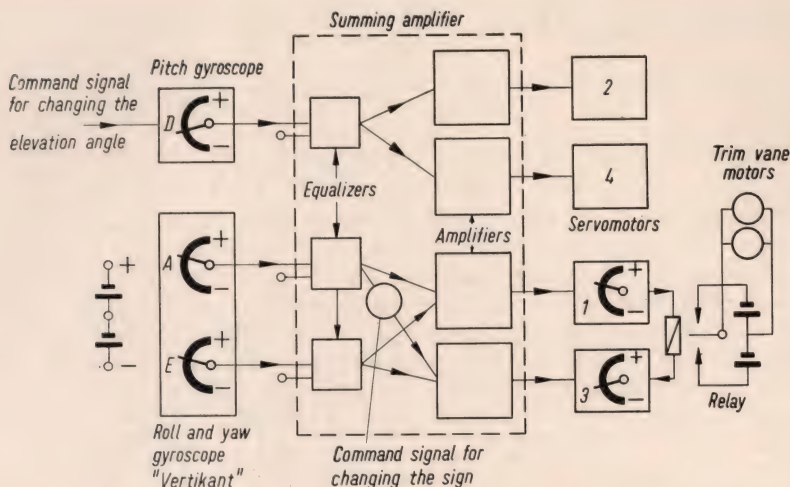


Fig. 3. Block diagram of control in yaw, pitch and roll

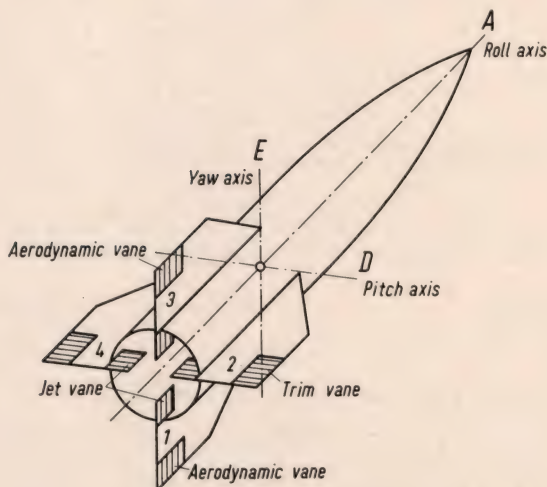


Fig. 4. Location of the vanes of the V-2

Fig. 4 shows the arrangement of the control vanes. The functioning of the trim vanes destabilizes the control in roll, but only to a permissible degree.

Special care has to be taken with regard to the correct position of the assigned course during launching and flight. In order to satisfy the first condition, the gyros were mounted on a kind of adjustable platform. The rocket was adjusted so that the platform was in a horizontal position, and one of its edges pointed towards the target (optical adjustment by collimator) (see Fig. 5). To make sure of the second condition, gyros were used, which in the laboratory did not wander more than 0.1 to 0.2 degrees per minute under the influence of gravity.

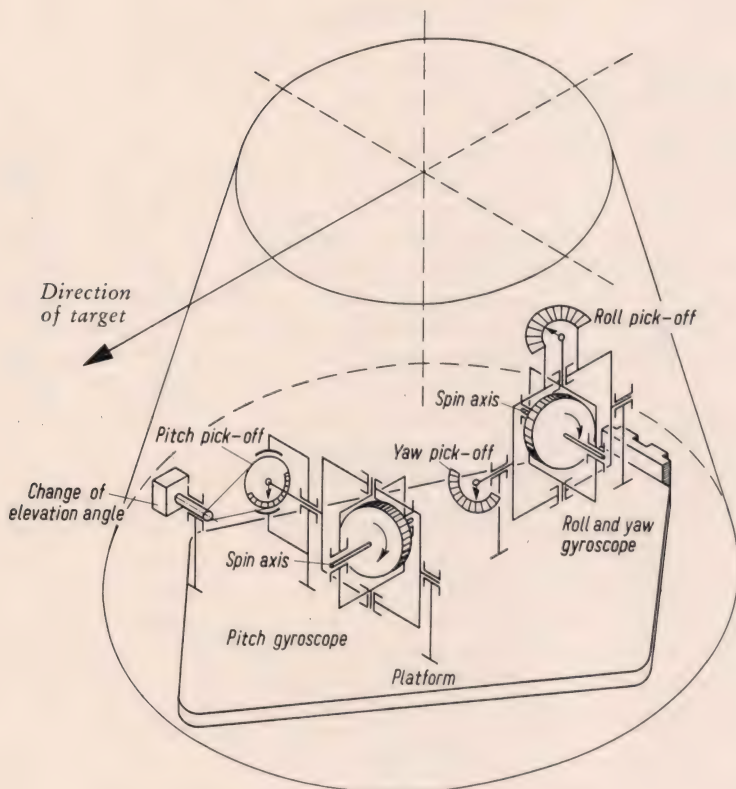


Fig. 5. Arrangement of the gyroscope in the missile

## 2.2. Fuel Control

### 2.2.1. General

For the range of the rocket we have as first approximation

$$(2.1) \quad X = f(v_1) \quad (v_1 = \text{fuel cut-off or all-burnt velocity}).$$

The fuel cut-off unit had to shut down the fuel supply when the rocket had reached the velocity assigned  $v_1^*$ . It consisted as is shown in Fig. 6 of,

- a device for measuring the rocket velocity, furnishing a quantity  $B$  proportional to this velocity near the cut-off point;

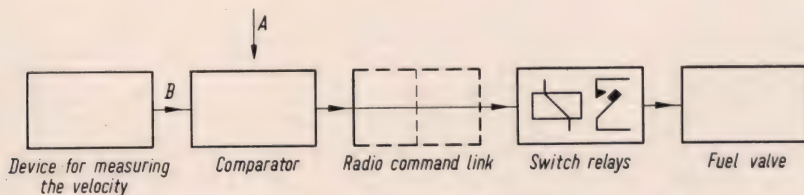


Fig. 6. Block diagram of the device for propulsion cut-off

- b) a device (comparator) releasing a signal when  $B$  exceeded a predetermined quantity  $A$ ;
- c) a relay operated by this signal which shut down the supply of fuel to the combustion chamber.

If the velocity measuring equipment was stationed on the ground an additional radio transmitter unit was required.

The quantity  $A$  is approximately proportional to the all-burnt velocity  $v_1^*$ . However, since the rocket velocity still increases in the time between the response of the comparator and the complete cessation of combustion,  $A$  was actually chosen to be proportional to a velocity which was smaller than the all-burnt velocity assigned by the mean value  $\Delta v_1^*$  of this increase  $\Delta v_1$ . Since the variance of  $\Delta v_1$  was relatively large,  $\Delta v_1$  was reduced by reducing the thrust immediately before cut-off in the ratio 4:1. This produced a thrust-time curve as shown in Fig. 7. Further corrections of  $A$  will be discussed later.

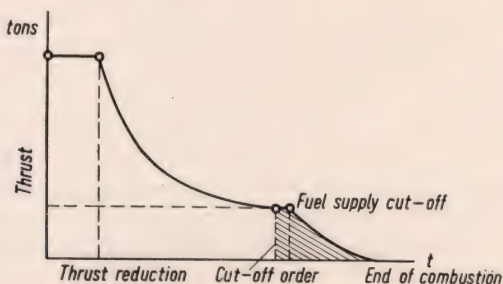


Fig. 7. Thrust at cut-off  
Hatched area = velocity increase  $\Delta v_1$

### 2.2.2. Description of the Two Systems

For the V-2 two different fuel shut-down systems were developed:

- a) Utilisation of DOPPLER-Effect.

From the ground, a measuring frequency (meter wave length) was transmitted to the rocket and from there, after doubling the frequency, reflected to the ground. The beat frequency  $B_t$  of the second harmonic of the measuring frequency and the reflected wave is proportional to the rocket velocity in the direction of the straight line joining the ground base with the rocket. The comparator contained a WIEN bridge which was tuned to a frequency  $A_t$  corresponding to  $v_1^* - \Delta v_1^*$ . When the beat frequency passed through this tuned frequency, a signal was transmitted to the rocket which operated the relays. In order to measure the rocket velocity itself, transmitter and receiver had to be positioned in the direction of the trajectory tangent at the instant of all-burnt.

- b) Integration of the acceleration of the rocket.

Here one tried to measure the acceleration of the rocket  $dv_t/dt$  in the direction of the tangent of the trajectory, and the ascertained value was integrated. Because of the gravitational acceleration one obtained the value

$$(2.2) \quad v_s = \int_0^t \frac{dv_t}{dt} dt + \int_0^t g \cos \vartheta dt,$$

where  $\vartheta$  is the angle of inclination of the trajectory.

The first integral represents the rocket velocity, and therefore the second term has to be calculated beforehand for the mean interval  $t_1^*$  between start and all-burnt, and then fed into the comparator

$$(2.3) \quad A_i \sim v_1^* + \int_0^{t_1^*} g \cos \vartheta dt - \Delta v_1^*.$$

The correction is, as was stated above, only correct in the average. Another inaccuracy is unavoidable; the accelerometer can be installed in the direction of the rocket axis but not of the trajectory tangent. But considering the small angle of attack, the error thus caused is only small.

Two different integrating devices were employed. The first (IG 1) utilized the fact that a gyro under the influence of an acceleration precesses at a rate proportional to the acceleration. The angle of precession  $B_{iw}$  was proportional to the integral with respect to time of all accelerations, i.e. the value  $v_s$ . The fuel cut-off signal was released when the gyro, reaching the angle of precession  $A_{iw}$ , closed a properly adjusted contact. The second device (IG 2) determined the velocity by means of a rotating pendulum fettered by an electric current. When the pendulum moved from its zero position under the influence of an acceleration, a bridge became unbalanced (Fig. 8). Thus a

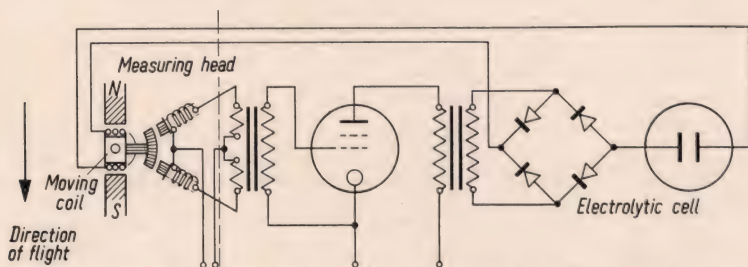


Fig. 8. Scheme of the electronic device for integrating the velocity of the missile

current was caused to flow in the moving coil which fettered the pendulum. The condition of equilibrium was such that the torque caused by the acceleration and the torque of the moving coil compensated each other. The moving-coil current was also proportional to the acceleration. This current was integrated by means of an electrolytic cell especially developed for this purpose. The charge  $B_{iL}$  of the cell during the flight is proportional to the velocity  $v_s$ . Before launching the cell was charged with  $A_{iL}$  in the opposite sense and caused fuel cut-off when the charge  $B_{iL} - A_{iL}$  passed through zero.

Because of the required gravity correction, the IG-method is less accurate than the radio method. The latter was to be preferred for another reason. By a minor modification of the base  $M$ , the influence of errors of the trajectory tangent angle  $\vartheta_1$  on the range  $X$  can be eliminated. The equation

$$(2.4) \quad X = f(v_1, \vartheta_1)$$

is more accurate than equation (2.1).

If  $\vartheta_1$  is slightly different from its assigned value  $\vartheta_1^*$ , a velocity must be chosen which, in its turn, is also slightly different from its assigned value in order to obtain its correct range. Again we have

$$(2.5) \quad X = f(v_x, v_y) \quad (v_x = v_1 \cos \vartheta, v_y = v_1 \sin \vartheta).$$

If  $v_x$  and  $v_y$  differ only slightly from their assigned values, the following equations holds:

$$(2.6) \quad X = \frac{\partial f}{\partial v_x} v_x + \frac{\partial f}{\partial v_y} v_y + E \quad (E = \text{a numerical value}),$$

Formula (2.6) can be transformed into

$$(2.7) \quad X = \sqrt{\left(\frac{\partial f}{\partial v_x}\right)^2 + \left(\frac{\partial f}{\partial v_y}\right)^2} v_a + E$$

with

$$(2.8) \quad \cot a = \frac{\partial f / \partial v_y}{\partial f / \partial v_x}.$$

Thus a given range depends only on the rocket velocity in the direction  $a$ . We eliminate the influence of the angle of the trajectory tangent if we position the base  $M$  so that the straight line joining  $M$  and the rocket at the time of cut-off contains the angle  $a$  (instead of  $\vartheta_1^*$ ) with the vertical. A corresponding correction is not possible with an integrating device fixed to the axis of the rocket. The above partial derivatives were obtained from trajectory calculations.

### 2.3. Some Trial Results

For the average deviation, quoting from memory, we obtained the following results:

Error in line	$z_{50} = \pm 4.5 \text{ km}$
Error in range with cut-off by IG-method	$x_{50} = \pm 4.5 \text{ km}$
Error in range with cut-off by radio	$x_{50} = \pm 4 \text{ km}$

It should however be said that the last IG-method results were somewhat better than the last radio-method results. The reason for this somewhat surprising result is not known with certainty. We conjectured the following: the range depends also on the position of the rocket at fuel cut-off. If now the thrust (especially after its reduction) is too weak, the rocket travels for a longer time and therefore over a greater distance than originally intended, until it reaches the fuel cut-off velocity. In the case of the radio-method, this causes an error in range only, whereas with the IG-method an error in time is introduced due to the influence of gravity. This error, as can be seen from equation (2.2),

operates in the opposite direction. Whether this compensation actually improved the accuracy noticeable is not quite clear.

Part of the observed deviation is naturally caused by the wind and other influences during the time of free flight. The error thus caused was estimated at a few hundred meters.

The influence of the rotation of the earth on the trajectory (some kilometres) made the use of range tables necessary.

### 3. THE GUIDE PLANE CONTROL OF THE SPECIAL SERIES

#### 3.1. Description

As already mentioned, the rocket, if controlled by the automatic pilot only, will drift laterally under the influence of lateral forces. On an average this causes a considerable deviation. If, at the instant of fuel cut-off, the rocket has a velocity of say, 1400 m/sec corresponding to an approximate range of 250 km and the lateral wind has an assumed velocity of 20 m/sec, an error in line of about 3.5 km is to be expected. Further noticeable lateral deviations must be expected as a result of the wandering of the course gyro, and asymmetries in thrust. In order to reduce these inaccuracies, some of the V-2 rockets were equipped with guide-plane control. It had been demanded that this control should operate through an additional signal to the automatic pilot. It was intended to operate the rockets with automatic pilot only in the case of interference. Furthermore production problems caused that demand. It may be mentioned here that the radio connection to the V-2 was never jammed.

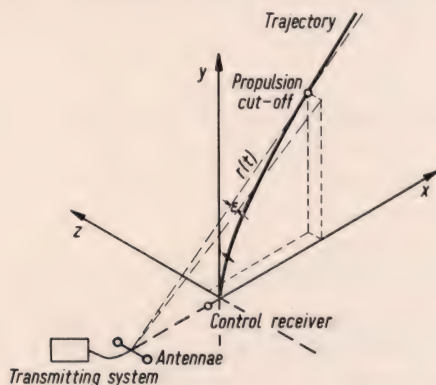


Fig. 9. The radio range control of the V-2

The guide-plane was produced as follows (Fig. 9): A transmitter with a frequency of about 50 mc/sec was positioned at a distance of 10 km behind the launching base, so that transmitter and launching base were contained in the plane intended as guide-plane. The transmitter fed two horizontal dipoles positioned at a distance of about 200 metres from each other. The feeding into the connecting line of the two aerials was effected alternately at two points which had the same distance from the nearer aerial and the distance of about

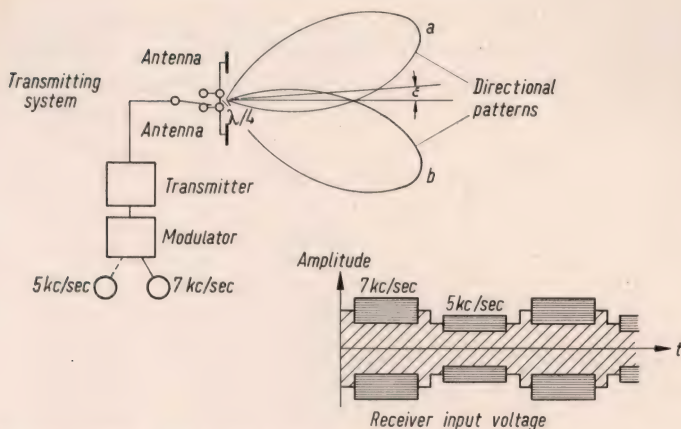


Fig. 10. Transmitting system of the radio control system with directional patterns; input voltage to the receiver

$\lambda/4$  from each other (Fig. 10). The switching frequency was 50 c/sec and beam characteristics *a* and *b* were obtained alternately (the parasitic lobes are suppressed). At the receiver input one obtained a square modulated voltage whose depth of modulation was very approximately proportional to the angle  $\epsilon$  formed by the straightline transmitter—receiver with the guide-plane. An additional modulation of the transmitter of 7 kc/sec made it possible to find out whether the receiver was to the right or to the left of the guide-plane.

The position of the guide-plane was monitored by a control receiver arranged in the desired direction of the guide-plane. At the receiver output in the rocket a voltage was obtained whose positive or negative sign depended on the lateral position (right or left) of the rocket and was proportional to its angular position  $\epsilon$ . This voltage was, via a stabilizer network, fed into the amplifier for the control in yaw. This network formed beside the  $\epsilon$  signal also an isodrome control because, in the case of a pure angle control (including its derivatives), the rocket, under the influence of lateral forces or of a gyro deviation, travels at a certain distance parallel to the guide-plane. The isodrome signal destabilized the rocket and therefore had to be chosen sufficiently weak. The guide-plane system was, like all other radio equipments for the V-2, developed for ten frequencies in order to dodge interference.

### 3.2. The Control Equations for the V-2 Automatic Pilot and Yaw Control with the Guide-Plane

Now we shall consider the problem of finding a suitable form of control equations. The three dimensional motion of a rocket with its six degrees of freedom can be described by six simultaneous differential equations of the second order. The coupling of these equations is considerably reduced if the rocket — like the V-2 — does not roll (this was another reason for aiming at a good stability in roll). It was found that the remaining coupling between pitch and yaw control could be neglected for a rocket with the aerodynamic characteristics and the trajectory of the V-2, so that the motion of the V-2 could be considered separately in each plane.

The stability problem is nearly identical for yaw, pitch and roll. Therefore we shall consider in detail how the equation for control in yaw was obtained, because in this case the additional problem of lateral control by guide-plane had to be dealt with.

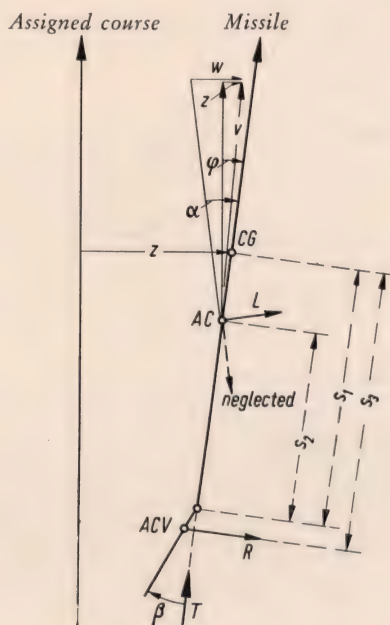


Fig. 11. Diagram of the forces which enter in the yaw control

GG = Centre of gravity, AC = Aerodynamic centre, ACV = Aerodynamic centre of vanes

Fig. 11 shows the lateral forces. The equation of motion are with good approximation if we take into consideration the little damping forces:

$$(3.1) \quad \Theta \frac{d^2 \varphi}{dt^2} + \delta^* \frac{d\varphi}{dt} + \frac{\partial L}{\partial \alpha} (s_1 - s_2) \alpha = - \frac{\partial R}{\partial \beta} s_3 \beta,$$

$$(3.2) \quad m \frac{d^2 z}{dt^2} = \frac{\partial L}{\partial \alpha} \alpha + \frac{\partial R}{\partial \beta} \beta + D^* \frac{d\varphi}{dt} + T \varphi,$$

where

$\Theta$  = moment of inertia,

$L \approx \frac{\partial c_l}{\partial \alpha} S \frac{\rho}{2} v^2 \alpha$  = lateral force,

$c_l$  = lateral force coefficient,

$S$  = cross-section of the rocket,

$\varphi$  = angle between longitudinal axis and assigned course of the rocket,

$$R = \frac{\partial c_{lr}}{\partial \alpha} S \frac{Q}{2} v^2 \beta = \text{controller force,}$$

$c_{lr}$  = controller force coefficient,

$\beta$  = controller angle,

$m$  = mass of the rocket,

$T$  = thrust,

$$D^* \frac{d\varphi}{dt} \approx \frac{\partial c_l}{\partial \alpha} S \frac{Q}{2} v^2 \frac{s_1}{v} \frac{d\varphi}{dt} = \text{damping force,}$$

$$\delta^* \frac{d\varphi}{dt} \approx \frac{\partial c_l}{\partial \alpha} S \frac{Q}{2} v^2 \frac{s_1^2}{v} \frac{d\varphi}{dt} = \text{damping moment,}$$

$s_1 - s_2$  = distance between the centre of gravity and the aerodynamic centre,

$s_3$  = distance between the centre of gravity and the aerodynamic centre of vanes.

The angle of attack is

$$(3.3) \quad \alpha = \frac{w - dz/dt}{v} + \varphi,$$

where

$w$  = velocity of lateral wind,

$z$  = lateral distance of the rocket from its assigned trajectory.

If  $d/dt$  is written as  $p$ , from (3.1), (3.2) and (3.3) follows

$$(3.4) \quad p^2 \varphi + \delta p \varphi + c_1 \varphi - k p z = -k w - c_2 \beta,$$

$$(3.5) \quad -p^2 z + D p \varphi + C_1 \varphi - K p z + C_2 \beta = -K w.$$

The suitable control equation can be written as follows if we assume that we are using a passive control network:

$$(3.6) \quad \left\{ \begin{aligned} \beta(m_0 + m_1 p + m_2 p^2 + m_3 p^3) &= \frac{1 + a_{10} p + a_{20} p^2}{1 + T_1 p + T_2 p^2} a_0 \varphi + \\ &+ b_0 \frac{1 + b_{10} p}{1 + T_3 p} z + b_0 \frac{b_{10}}{1 + T_4/p} \frac{z}{p}. \end{aligned} \right.$$

The first term on the right hand side signifies the gyro signal, the second the actual guide-beam signal, and the third the additional isodrome control. We still have to express  $z$  by the guide beam angle  $\varepsilon$ :

$$(3.7) \quad z = r(t) \varepsilon,$$

$$(3.8) \quad b_0 z = e_0 \varepsilon.$$

if we examine the problem of stabilization only.

It can be easily seen that the first term is suitable in the case of gyro control only. Equation (3.5) is to be omitted and equation (3.4) is simplified into

$$(3.9) \quad p^2 \varphi + \delta p \varphi + c_1 \varphi = c_2 \beta,$$

if we examine the problem of stabilization only.

For a position servo unit a good stabilization is obtained with the equation

$$(3.10) \quad m_0 \beta = a_0 \varphi (1 + a_{10} p)$$

and for a velocity servo unit with the equation

$$(3.11) \quad m_1 p \beta = a_0 \varphi (a_{10} p + a_{20} p^2).$$

For from (3.9) together with (3.10) and (3.11) respectively we obtain

$$(3.12) \quad p^2 \varphi + p \varphi \left( \delta + \frac{a_0 a_{10} c_2}{m_0} \right) + \left( c_1 + \frac{a_0 c_2}{m_0} \right) \varphi = 0,$$

$$(3.13) \quad p^2 \varphi + p \varphi \left( \delta + \frac{a_0 a_{20} c_2}{m_1} \right) + \left( c_1 + \frac{a_0 a_{10} c_2}{m_1} \right) \varphi = 0.$$

The oscillation about the centre of gravity is additionally damped in the first case by the first, in the second case by the second differential quotient of the control equation. The servo unit was originally rate-coordinated, however, the net hinge-moments of the control vanes actually make  $m_0$  somewhat different from zero. It is therefore probable that a signal of the form  $a_0 (1 + a_{10} p + a_{20} p^2)$  produces good stability. Naturally,  $a_0$  would have to be present in the signal even if it effected destabilization, since it determines the assigned course of the automatic pilot.

The terms in the denominator are unavoidable if a passive control network is used. In the final calculation of an optional transfer function for the signal, they naturally had to be considered; moreover, the coefficient  $m_2$  carried considerable weight. As mentioned above a network was developed according to equation (3.6), which permitted the control characteristics to be kept constant during the entire control period.

Based on similar considerations, it was at first assumed that the guide-plane signal had also to be formed by a double differentiating network. However, the calculations showed that single differentiation was sufficient. Since, in the case of the guide-plane control, the controlled motion of the rocket is represented by two simultaneous differential equations, the stability calculations become far more extensive than those for the automatic pilot. Moreover, they are less reliable because such calculations can be carried out in practice only for equations with constant coefficients. This is permissible for the automatic pilot because the oscillation about the centre of gravity is sufficiently rapid; however, in the case of the guide-plane control the values thus obtained appear rather questionable because the oscillation of the centre of gravity is very slow. It was found, however, that these calculations give quite a good idea of the stability of the guide-plane control. Control characteristics and transfer functions for the first guide-plane controlled rockets were determined by means of such calculations and produced satisfactory stability.

It was found that during the control period the transfer function could be kept constant, but that the amplification had to be varied. This was accomplished by a cam gear-driven-potentiometer at the input of the yaw control amplifier.

The fact that the end of the propelled part of the trajectory  $e_0$  had to be varied considerably, unfortunately destroyed to a great extent the effect of the isodrome control, which was also proportional to  $e_0$ . It was not possible to develop methods avoiding this disadvantage.

The denominator term of the guide-plane signal calls for the following remarks: while the denominator terms of the automatic pilot system signal were

chosen so small — they were regarded as undesirable — as was technically permissible (attenuation factor of the network)  $T_3$  was chosen as large as possible in order to filter the signal, but without jeopardizing the stability of the control: the possibility of interference was to be reduced. In the end, filtering was still carried further by a guide-plane signal of the type (approximately)

$$(3.14) \quad e_0 \frac{1 + b_{10} p}{(1 + T_3 p)^2} z.$$

In spite of many efforts we did not succeed in making the lateral control of the rocket really stable during the last seconds before all-burnt. Although the rocket always remained in close proximity to the guide-plane, we often observed an angle between the trajectory tangent at all-burnt and the guide-plane, which could not be neglected.

The systems of equations for the control in roll and pitch are somewhat different from those for the control in yaw, but the difference is not material and, therefore, we do not consider them here.

### 3.3. The Methods Used for Stability Control. — Numerical Methods and Analogue Computers

The numerical investigations of the stability problem were almost exclusively carried out by the method of small disturbances, i.e. the roots of the characteristic equation of the system of differential equations were examined. For this purpose, the method of examining the determinant of the highest order did not appear to be particularly suitable; instead, damping régimes were calculated specially, by a method using the conditions for the coefficients. The results of this method may be explained in a few words: if for instance the other coefficients of the guide-plane control are given, the limits are obtained with which  $b_0$  and  $b_{10}$  may be chosen, if the damping of the oscillation should not be less than a certain value. In principle, the method requires a considerable amount of calculation; however, approximate methods were developed to reduce the labour.

Investigations into the stability were also carried out with the help of the NYQUIST method, but this was not so popular.

The stability of the pure automatic pilot control was also investigated with the help of an electro-mechanical analogue computer, because it was not possible to consider the backlash of the servo-unit and the non-linearities in the calculation. The computer was a kind of moving-coil voltmeter (Fig. 12) whose pointer had been enlarged to a pendulum. A small eddy current brake made it possible to adjust the damping of the oscillations of the pendulum. By tilting the pendulum in a plane perpendicular to its axis, its natural frequency could be altered. The following equation was obtained for the free oscillations of the pendulum:

$$(3.15) \quad m_p^2 p^2 \varphi + d_m p \varphi + m_p g r \cos \gamma \cdot \varphi = 0,$$

where

$m_p$  = mass of the pendulum.

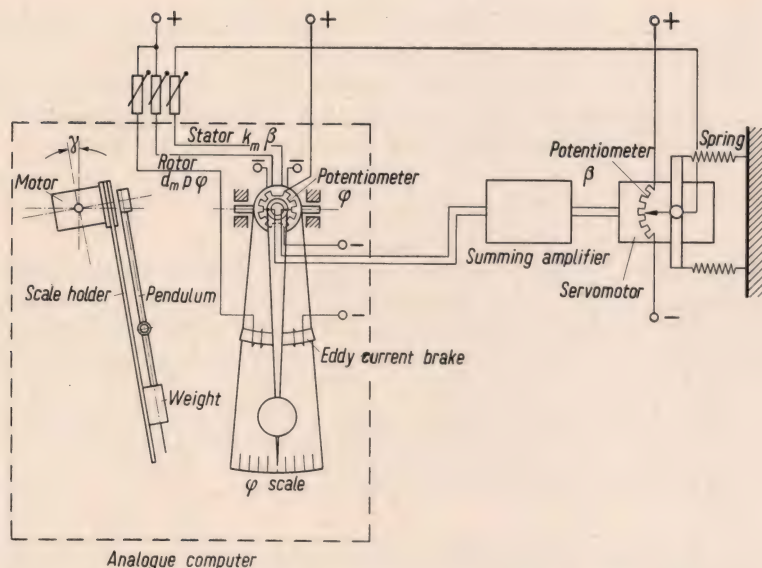


Fig. 12. Autopilot control simulation: block diagram

The design data were chosen in such a way as to give the pendulum the particular natural frequency and damping characteristic for the rocket in any point of the controlled part of its trajectory.

In order to investigate the stability of the control system a voltage was determined, with the help of a potentiometer, which was proportional to the angle  $\varphi$  between the axis of the pendulum and the vertical (the arrangement was such that this angle was identical with the angle  $\varphi$  between the axis of the rocket and its assigned course). This voltage was fed into a mixer unit which operated the servo-unit. The servo-unit carried a potentiometer which produced a voltage proportional to the deflection  $\beta$  of the controller this voltage exercised a proportional torque on the pendulum. The hinge-moment set up at the servo-unit during flight was imitated by a spring coupling. According to the above the equations for the whole system were:

$$(3.16) \quad m_p p^2 \varphi + d_m p \varphi + m_p g r \cos \gamma \cdot \varphi + k_m \beta = 0,$$

$$(3.17) \quad m_0 \beta + m_1 p \beta + m_2 p^2 \beta = a_0 \varphi \frac{1 + \dots}{1 + \dots},$$

this being identical with the system of equations for the pure automatic pilot control. It was observed how the system returned to zero after a disturbance.

The imitation of the system of equations for the guide-plane control seemed to be practically impossible in an electro-mechanical analogue computer, especially because the coefficients had to be variable with time. Therefore an electronic analogue computer had to be developed for these stability investigations. This contained essentially a number of integrators and potentiometers operated by cam shafts. It was possible with the help of these potentiometers to reproduce the variation with time of the coefficients of the system of equations.

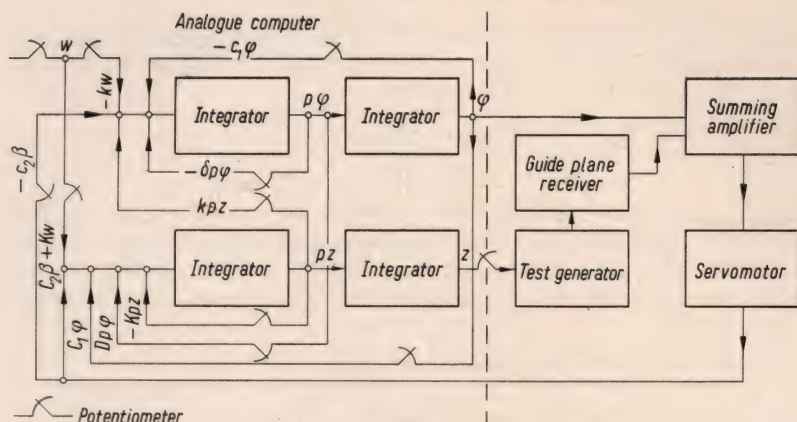


Fig. 13. Radio control simulation: block diagram

Fig. 13 shows the basic diagram of the computer. One recognizes at once that the input voltage of the two integrators on the left is  $p^2 z$  and  $p^2 \varphi$ , if one approaches the input via the integrator; by adding the voltage fed into the inputs we see that yields:

$$(3.18) \quad p^2 \varphi = -\delta p \varphi - c_1 \varphi + k p z - k w - c_2 \beta,$$

$$(3.19) \quad p^2 z = D p \varphi + C_1 \varphi - K p z + K w + C_2 \beta.$$

The analogue computer thus imitates the equations of the motion of the rocket. The control equation (3.6) is represented by the control elements themselves.

It should be mentioned that the servo-unit was loaded by a leaf spring whose point of support was varied with time in such a way as to imitate the hinge-moment correctly at any time.

Between the output of the analogue computer and the input of the guide-plane receiver which, according to section 3.1, supplies at its output the sum of the actual guide-beam signal and of the isodrome signal was installed a specially developed control generator. By the voltage  $\varepsilon = z/r(t)$  this generator was modulated so that at its output it delivered a voltage curve corresponding to the one occurring at the input of a receiver placed at the angular distance  $\varepsilon$  from the guide-plane (see Fig. 10).

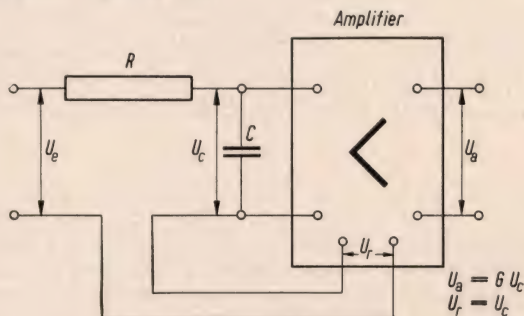


Fig. 14. Principle of an integrating amplifier

The integrators consisted of capacitors which were charged via resistances. Fig. 14 shows the principle. The condenser voltage is fed-back into the input circuit via a buffer amplifier. Thus we have

$$(3.20) \quad U_a = \frac{1}{pC} \frac{U_e}{R} G$$

as long as one operates on the linear part of the amplifier characteristic. If the amplification of the feed-back-loop is somewhat different from 1 the integration is no longer ideal; however, it was possible without great difficulty to keep the time constant greater than ten minutes for half an hour if the resistance  $R$  was not too small.

In order to avoid the difficulties which usually occur with a number of D.C. amplifiers in series, the inputs and outputs of the integrators carried 500 c/sec voltages. The input voltages were summed up, the total voltage rectified with a phase-sensitive detector, then the integration carried out and finally the integrated value used to modulate at 500 c/sec suppressed carrier system (see Fig. 15). The feeding back as described above was carried out with 500 c/sec from output to input of the integrator.

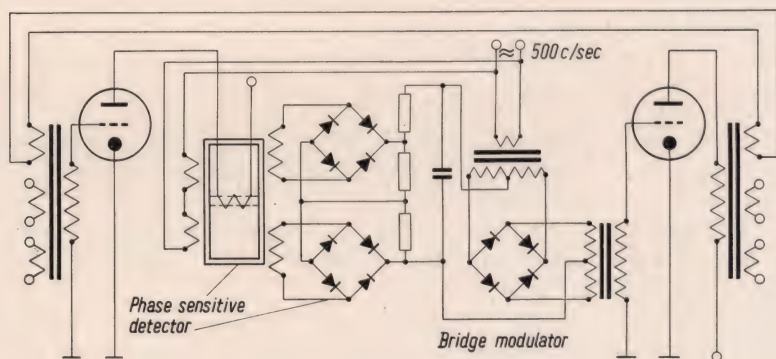


Fig. 15. Simplified circuit of an integrating amplifier

The operation of the computer was made rather difficult by the necessity of changing a cam shaft if the time function of the coefficient was to be altered. It was therefore intended to equip the potentiometers with step-by-step selectors and to operate these with the help of perforated tape.

The investigations carried out with the electronic analogue computer were of great importance, especially for the solution of the problem of finding the optimum control characteristic and the optimum transfer function for the guide-plane control.

### 3.4. Trial Results

The probable error of the rockets with guide-plane control was within one or two kilometres. This error was caused mainly by the angle between the direction of the trajectory tangent at all-burnt and the guide-plane (deviation of  $0.1^\circ$  causes in the case of a range of 250 km a lateral deviation of about 400 m). We were convinced that the probable error would be considerably below one

km if the problem of improving the damping of the control immediately before all-burnt could be solved.

#### 4. HIGH PRECISION CONTROL WITHOUT RADIO

##### 4.1. Description

The integration units had been developed for a V-2-control without radio. But a combination of the described control system with a fuel cut-off unit containing an integrator fixed to the rocket axis could not be expected to produce good accuracy, for the following reasons:

- a) The gyros precessed too much under the influence of the considerable accelerations.
- b) The integration of the axial acceleration, and the use of the integrated value for the cut-off signal is, as we have seen, not very advantageous. — Moreover, we had no idea how to supplement this control system by an additional lateral guidance.

In order to remedy this, the following control system without radio was developed: A stabilized platform in Cardan suspension was built into the rocket. By means of three gyros this platform did not alter its direction in space. The angles between the three rocket axes and the platform were used as control quantities for the automatic pilot. Such a platform has a much better stability than a single gyro.

The pick-up of the integrator was mounted on this platform in such a way as to measure the component of the rocket acceleration in the direction  $a$ , which was used as measuring direction for the measurement of the rocket velocity by radio. Thus the determination of the fuel cut-off time by means of the integrator becomes theoretically equivalent to its determination by means of radio measurement. Only objection: the term for the correction of gravity which takes the form of

$$(4.1) \quad v_g = g \cos a \cdot t_1^*$$

is exact only if the time  $t_1$  up to all-burnt does not differ from its value  $t_1^*$  used in the calculation.

For additional lateral guidance — substitute for guide-plane control — a so-called lateral integrator was developed. An acceleration pick-up was mounted on the stabilized platform in such a way as to measure the acceleration perpendicular to the plane of the trajectory. The pick-up produced a current proportional to this acceleration. A network which in principle performed a double integration of this current yielded the expression

$$(4.2) \quad b_0 z (1 + b_{10} p)$$

which was used for the control instead of the guide-plane signal. It was not intended to use an integral signal for the time being. The probable error of the firings without additional guidance (only a limited number was available) amounted to

$$x_{50} = \pm 3.4 \text{ km,}$$

$$z_{50} = \pm 2.7 \text{ km.}$$

These rockets were equipped with the electronic integrator. The additional lateral guidance by means of the lateral integrator could only be tried once. The result was a lateral deviation of 0.5 km.

#### 4.2. Principle of Stabilized Platform and Lateral Integrator

Fig. 16 is a basic diagram of the stabilized platform. Each of the three gyros has one degree of freedom. If, due to a rotation of the Cardan suspension, a friction torque becomes effective, the respective gyro begins to precess and exerts on the platform a torque of equal magnitude in the opposite direction.

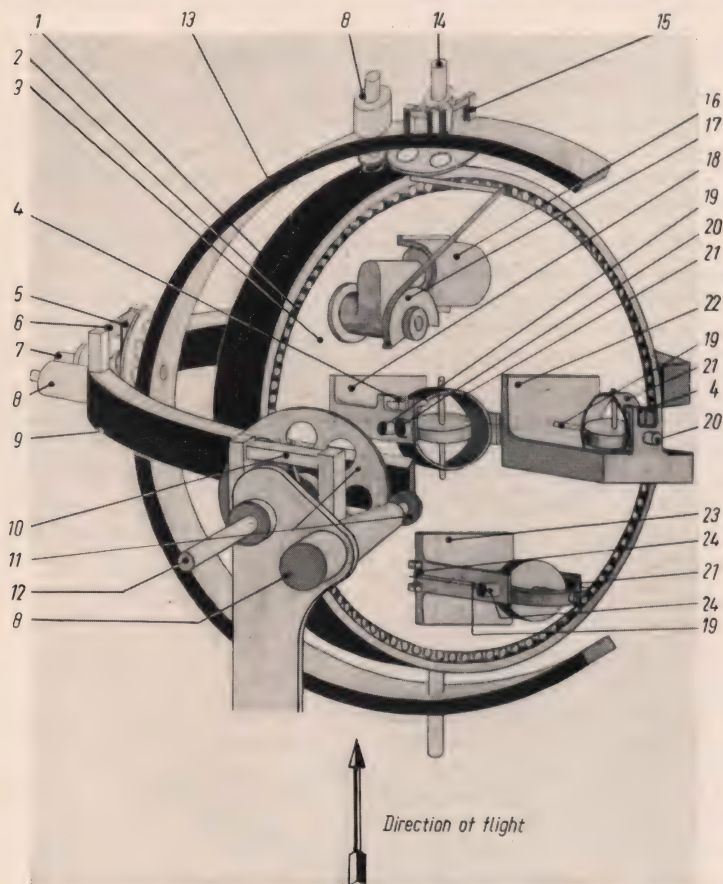


Fig. 16. Sketch of the stabilized platform

- |                           |                       |                      |
|---------------------------|-----------------------|----------------------|
| 1 = Bearing ring          | 2 = Wire ball bearing | 3 = Stable platform  |
| 4 = Contact of precession | 5 = Slider            | 6 = Yaw pick-off     |
| 7 = Yaw axis              | 8 = Torque motor      | 9 = Outer gimble     |
| 10 = Pitch pick-off       | 11 = Gear train       | 12 = Pitch axis      |
| 13 = Inner gimble         | 14 = Roll axis        | 15 = Roll pick-off   |
| 16 = Program motor        | 17 = Curved disc      | 18 = Pitch gyroscope |
| 19 = Precession axis      | 20 = Torque magnet    | 21 = Gimble          |
| 22 = Yaw gyroscope        | 23 = Roll gyroscope   | 24 = Magnet          |

The platform retains its direction in space. If the gyro has precessed through a certain angle it closes a contact supplying current to the respective supporting motor. This motor then exerts a torque on the platform greater than the possible friction torque. The gyro begins to vibrate around its precession axis. The platform too vibrates somewhat, but the amplitude of its vibration is negligibly small. For good stability of the platform it is essential that the gyros have very little friction about their precession axis.

The change of direction in pitch was carried out as follows: a motor rotated the platform relative to the outer casing of its needle bearings. Naturally, the Cardan suspension ring rotates and not the platform. The potentiometer  $D$  supplies a voltage to the servo-units and the rocket changes its direction until the original constellation of longitudinal axis of the rocket and the Cardan suspension ring has been restored.

The simple co-ordination of gyro and supporting motor on which our description is based, no longer holds in the case of the changes in direction, but this is not a fundamental difficulty.

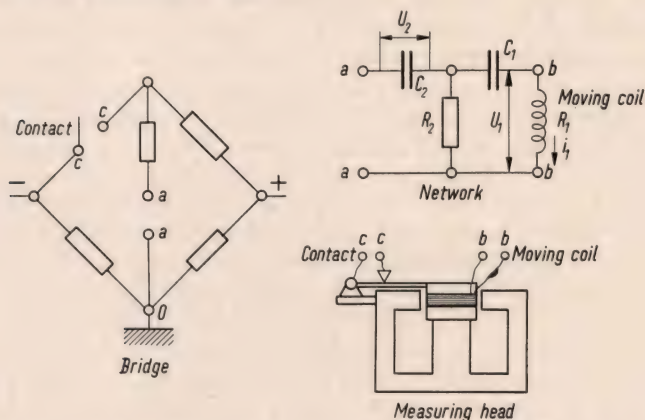


Fig. 17. The device for integrating the acceleration in yaw: sketch of the measuring head; principle of the bridge network

Fig. 17 shows the lateral integrator. A moving coil which dips into a magnet — the system was similar to a permanent dynamic loudspeaker — was built into the zero branch line of an emergency bridge circuit. Between the opposite points of the bridge and the moving coil is a network. One of the branches of the bridge is replaced by a switch which opens if the depth of the moving coil in the magnet exceeds a certain value. If the apparatus is switched on the coil vibrates with a relatively high frequency about the position thus determined. The mean value of the current in the coil is proportional to the acceleration exerted on the coil in the direction of its axis. Its polarity changes with the direction of the acceleration  $i_i \sim d^2 z / dt^2$ .

The voltage at the terminals  $aa$  of the network is

$$(4.3) \quad U_2 = \frac{1}{C_1 C_2 R_2} \iint i_1 dt \cdot dt + C_1(R_1 + R_2) \int i_1 dt$$

or

$$(4.4) \quad U_2 = k z [1 + p (C_1 R_1 + C_1 R_2)].$$

The voltage  $U_2$  thus contains a component proportional to  $z$  and another proportional to  $dz/dt$ . A signal was obtained similar to the one supplied by the guide-plane receiver, feeding  $U_2$  into an amplifier whose gain was varied by means of a cam shaft driven potentiometer. The output voltage of the amplifier was fed into the mixer unit.

A moving coil system pick-up would have given too large a systematic error. A component of the longitudinal acceleration proportional to the angle of deflection would have been included in the measurement. If a system had been chosen which avoided the vibration of the moving coil, friction would have been intolerable.

## 5. SPECIAL EQUIPMENT FOR THE CONTROL OF THE V-2 WHICH WAS IN COURSE OF DEVELOPMENT

### 5.1. The Guide-Line Control

To protect the radio yaw control better against interference and also in order to make a radio pitch control possible, i. e. a control in two planes vertical to each other, a guide-line system in the 50 cm band was developed.

A continuous wave-transmitter fed the antenna equipment of a radar set, i. e. a dipole rotating eccentrically within a parabolic reflector. The airborne receiver determined the angular distance between the pilot beam and the radius vector to the rocket as an altitude component  $\varepsilon_h$  and a lateral component  $\varepsilon_v$ . It was similar in operation to the receiver of the radar sets. Due to the horizontal polarization of its aerial, the output voltage of the receiver — making certain approximations — was

$$(5.1) \quad U_3 = k_I (2 \varepsilon_v \sin \omega t + \varepsilon_h \cos \omega t) + k_{II} \cos 2 \omega t,$$

where  $\omega$  = frequency of rotation of dipole. The term  $k_I$  is led to two phase-sensitive rectifier circuits, into which are fed additional voltages  $k_{III} \cos \omega t$  and  $k_{III} \sin \omega t$  respectively. At the output the circuits supplied directional voltages proportional to  $\varepsilon_h$  and  $\varepsilon_v$  in amplitude. The voltages  $k_{III} \cos \omega t$  and  $k_{III} \sin \omega t$  were obtained from  $k_{II} \cos 2 \omega t$  by frequency division. Since no experience was available as to how a guide-plane control device would behave when the guide-plane was being turned, the first lay-out of this set was planned so that the yaw control signal would control the rocket laterally, whereas the pitch control signal was transferred to the ground and this made the reflector follow the rocket vertically. This set was under trial, but has not yet been used for the actual control of a rocket. We were anxious to know whether the mechanical accuracy of the large 7 m parabolic mirror would be sufficient for following the change of direction without trouble.

A radio pitch control would have had the important advantage that the location of the stationary equipment on the ground would have become uncritical for the radio measurement of the velocity.

### 5.2. Additional Apparatus for Improving the Accuracy of the All-Burnt Signal

The range of the V-2 depended, apart from the all-burnt velocity and its direction, primarily upon the co-ordinates of the all-burnt spot  $x_1$  and  $y_1$ , which

normally deviate somewhat from their assigned values. For a better approximation we may replace equation (2.6) by the following relation:

$$(5.2) \quad X = x_1 + \frac{\partial f}{\partial y_1} y_1 + \frac{\partial f}{\partial v_x} v_x + \frac{\partial f}{\partial v_y} v_y + E_2$$

( $E_2$  = a numerical value). Transforming this equation as was done with equation (2.4), we get

$$(5.3) \quad X = k_1 v_a + k_2 r_\delta + E_2,$$

$$(5.4) \quad k_2 = \sqrt{1 + (\partial f / \partial y)^2},$$

$$(5.5) \quad \cot \delta = \partial f / \partial y.$$

Thus it is sufficient to measure the component  $r_\delta$  of the distance covered by the rocket in a direction inclined at an angle  $\delta$  with the vertical, and at an angle  $\theta$  with the relevant plane of yaw control, and to consider this component as the so-called "way-correction", when determining the all-burnt point. The electronic "longitudinal integration method" makes it possible to determine  $r_\delta$  in a simple manner: a second pick-up, which measures the acceleration in the direction  $\delta$ , is placed on the stabilized platform, and the resulting current is integrated twice.

The radio measurement supplied the way-correction with satisfactory accuracy provided a second receiver unit was placed not too far away from the launching base. Labour developments were under way to incorporate the way-correction  $k_2 r_\delta$  in both cut-off procedures.

With the radio cut-off procedure the way-correction factor was used to alter the frequency of the WIEN bridge. With the integrator cut-off procedure it was intended to unload additionally, by a current proportional to the twice integrated measurement current, the electrolytic cell, which caused fuel cut-off when the charge passed through zero.

The additional equipment necessitated by the introduction of the way-correction was rather bulky. Devices were developed, therefore, which introduced, instead of a way-correction, a time-correction into the switching-off circuits.

The deviation of the distance all-burnt point-launching base from its calculated value was approximately proportional to the difference between the time required for the flight and its assigned value  $t_1 - t_1^*$ .

Instead of (5.3) we can, therefore, write approximately

$$(5.6) \quad X \approx k_1 v_a + k_3 (t_1 - t_1^*) + E_3.$$

With regard to the radio cut-off procedure, the frequency of the WIEN bridge was changed by a synchron-motor which began to rotate at the moment of launching. The first firing tests with this arrangement yielded satisfactory results.

With regard to the integration procedures, it was intended to change the position of the switching contact (IG 1) by a synchron-motor so as to send a constant additional current through the electrolytic cell (IG 2). These developments were not very far advanced, although a time-correction would have considerably reduced the apparent velocity compensation error which is also proportional to  $t_1 - t_1^*$ .

# DEVELOPMENT AND FIELD-TESTS OF A RADIO-CONTROLLED AEROPLANE MODEL AS TARGET SIMULATOR FOR ANTI-AIRCRAFT BATTERIES

WALTER KLOEPFER \*

## 1. INTRODUCTION

My purpose in this paper is to review the development of a miniature, radio-controlled light anti-aircraft target aeroplane. The proposal was first put forward by the ARGUS Motor Company which, on the initiative of Dr. GOSLAU, in 1936 began work on guided missiles. In April 1937 ARGUS asked the firm of LORENZ to undertake the development of the control system, and the DEUTSCHE FORSCHUNGSANSTALT FÜR SEGELFLUG (DFS) was asked to co-operate in the design of the airframe. The various activities were co-ordinated by ARGUS, and the work was given the title "ARGUS-Flak-Ziel-Modell".

The specification called for the construction of a simple, robust miniature aeroplane, capable of taking off from, and landing on, normal airfields. The dimensions were determined largely by the intended purpose, but a wing span of 2.4 m and a level speed of 90 km/h were chosen as most suitable. It was propelled by an ARGUS 3 h.p. single cylinder two-stroke engine with battery ignition. Since the control system was to be installed in the tubular body, the weight and number of components had to be kept as small as possible. Following the trend of the period, dry batteries were avoided and power was supplied from a generator.

Six orders were transmitted by the radio control system; two for the ailerons, two for the elevators and a fifth to control the ignition. The last order was carried as a spare and was sometimes used to operate a camera shutter. The orders were transmitted from the ground by means of a control stick, and it was possible to transmit two at a time. There was only a fraction of a second delay between the giving and the execution of an order.

## 2. PRINCIPLE OF THE RADIO CONTROL SYSTEM

The choice of transmission system was determined by the need to keep the cost of the receiver, decoder, control system and power supply to a minimum. Since the specification only asked for control within optical range of the control tower, 50 Mc/sec V.H.F. was selected for the radio transmission. A low cost

\* Dr.-Ing. — C. Lorenz AG., Pforzheim, Germany, Chief of Development, formerly of wireless sets for aeroplanes, at present of small wireless sets.

receiver was obtained by using super-regeneration together with the proper audio gain. The main advantages of this scheme were the ideal gain control characteristic and the high sensitivity. To avoid interference from the ignition and outside sources, a ground station with the rather high power of 40 W was used. A six-channel single tone system was used for the radio transmission. The six signals were assigned to six audio frequencies between 150 c/sec and 450 c/sec and were transmitted from the ground station when actuated by the control stick. The audio frequencies were filtered in the receiver by means of resonance relays and operated the control surfaces through intermediate relays. Considerable difficulties arose in the search for a suitable means of operating the controls, and attempts to use existing small servomotors failed because of the expense and long delays involved.

The first really workable solution was found by ARGUS, who proposed a pneumatic system which utilised the low pressure in the engine crank-case. In this system the controls were operated by a pressure cylinder connected to the engine crank-case through relay operated valves. Originally, a simple black-white signalling system, in which a deflection of the control stick produced full deflection of the corresponding control surface, was used, but the first flight tests showed that this made the course a little unsteady. An obvious improvement was to use pulses of variable duration in the control, and to deflect the controls in a smooth manner between zero and full deflection. At a later stage it was proposed to introduce the so-called pulse-ratio method.

### 3. TECHNICAL DETAILS OF THE AIRCRAFT EQUIPMENT

A view of the complete aeroplane is shown in Fig. 1. The mechanical structure was rather simple. The wings, tail unit, undercarriage and motor were attached to the tubular aluminium body by means of clamps, which made adjustments during assembly simple. The cylindrically-mounted receiver was put in from the rear of the tubular body.

Longitudinal stability was achieved by dihedral on the wings. The fabric covered wings had tubular aluminium mainspars and wooden ribs, and the tail unit was a flat fabric-covered steel tube frame. Both ailerons and elevators were fitted. The ailerons were operated by means of a control cylinder within the wings, and the elevators by a double-acting control cylinder in the rear

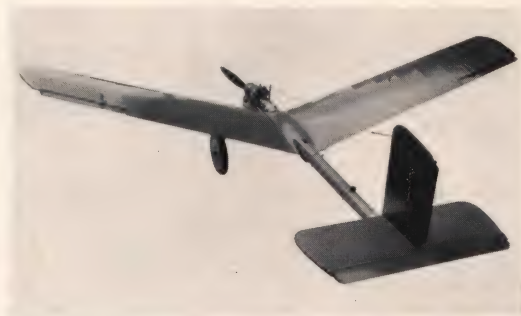


Fig. 1. Radio controlled target simulator. Assembly view

fuselage. The aeroplane was trimmed for straight and level flight, with all controls at neutral, by means of an additional tab on the tail unit.

The undercarriage consisted of two well sprung wide track main wheels and a tail wheel. The all-up-weight was 27 kg.

The engine assembly is shown in Fig. 2. The engine was a 70 cm<sup>3</sup> two stroke developing 3 h.p. at 6000 r.p.m. Ignition was obtained from a coil and torch battery, and the whole propulsion unit could be adjusted in both horizontal and vertical directions. Special care was taken to suppress the engine noise and it was necessary to completely screen the ignition system in order to reduce interference to a tolerable level. Since low weight dry batteries were not available, power was provided by an a.c. generator driven by the engine. It

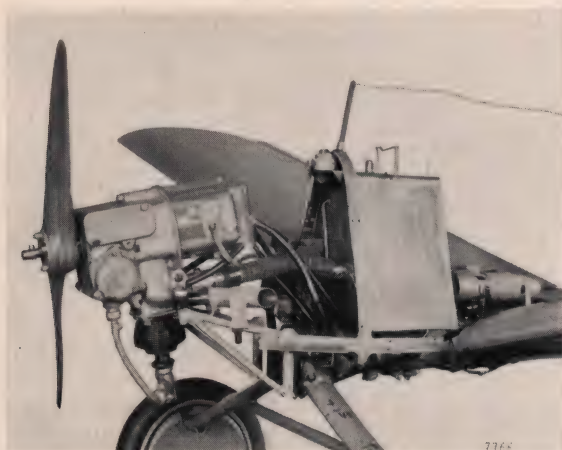


Fig. 2. Radio controlled target simulator. Propulsion gear with flanged generator

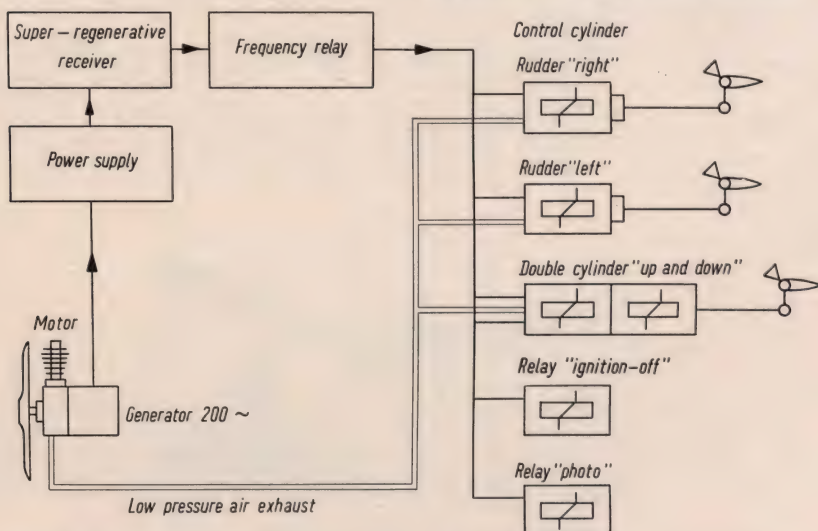


Fig. 3. Radio controlled target simulator. Block diagram of airborne equipment

delivered the necessary supply voltages for the filaments, high tension and relay actuation.

The basic circuit diagram of the aircraft equipment is shown in Fig. 3. The equipment consisted of the receiver proper, together with the resonance relays and the power supply rectifier. On the right hand side, a schematic arrangement of the pneumatic control system and the low pressure line to the engine crank-case is shown.

The inter-connection between the resonance relays and the control cylinders is shown in Fig. 4. A working relay was mounted within the control-cylinder,

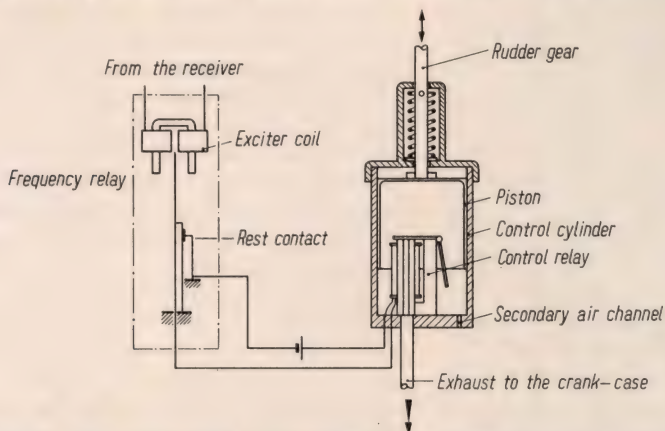


Fig. 4. Radio controlled target simulator. Principle of rudder actuation

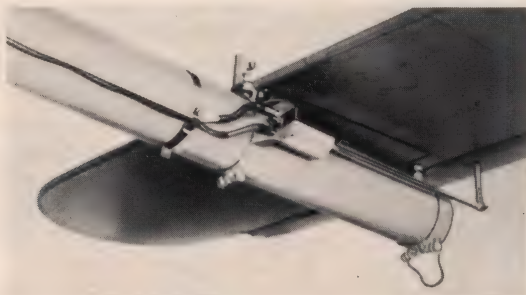


Fig. 5. Radio controlled target simulator. Control cylinder

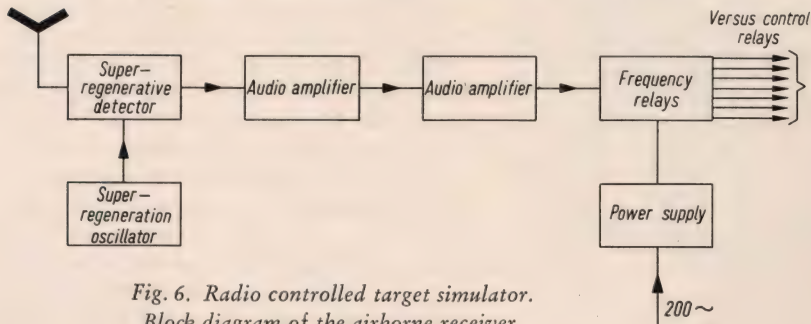
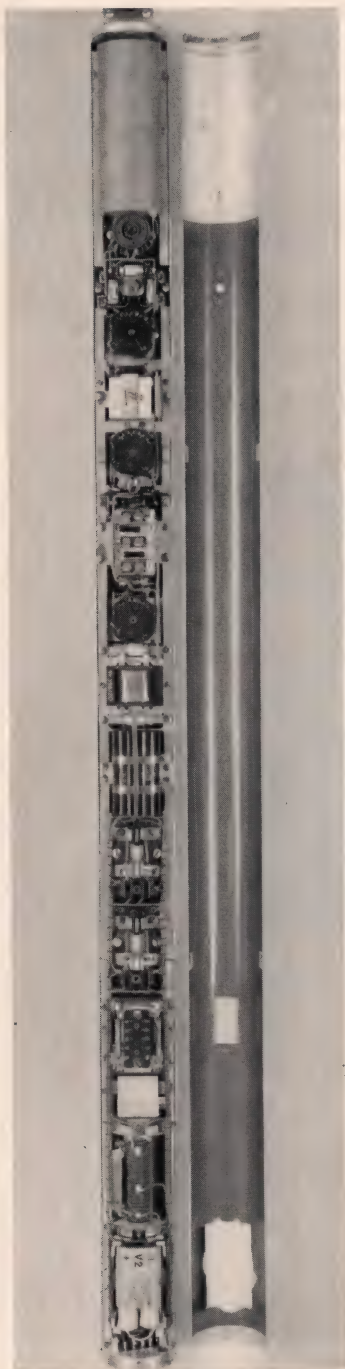


Fig. 6. Radio controlled target simulator.  
Block diagram of the airborne receiver



*Fig. 7. Radio controlled target simulator. Receiver inside*

which consisted of a cylinder and piston. One leg of this relay was hollow and was connected by a hose to the crank-case; so long as it was energised, the armature closed the opening of this leg and the piston stayed in the equilibrium position. But when a signal of the resonant frequency of the reed arrived from the receiver, the reed began to vibrate and periodically opened the rest contact. This released the armature of the working relay, opened the intake, exhausted the air from the cylinder, and the piston, moved under the outer air pressure. A spring returned the piston to its original position. In this way it was possible to produce control forces of several hundred grams, in an operating time of not more than a few tenths of a second. Since the armature did not completely close the air intake, some air was exhausted continually, and had to be compensated by a secondary air channel. This piston control operated on a large power margin and was therefore very reliable. The pneumatic control had the further advantage that, because of the balancing properties of the control cylinder, short interference signals which operated the primary relay had no effect on the controls.

Fig. 5 shows a preliminary model of the control cylinder.

Fig. 6 is a block diagram of the receiver. It consisted of a super-regenerative detector stage, with a separate oscillator for the regeneration frequency, and the audio amplifier stages. Rv 12 P 2000 valves were used throughout. A few  $\mu$ V of modulated RF at the receiver input were sufficient to operate the resonance relays, and because of the well-known gain control characteristic of a super-regenerative receiver, the output voltage of the resonance relays was largely independent of the distance between the model and the ground transmitter.

For reasons of stability, the six resonance relays were mounted within one sub-unit. A pair of reeds was assigned to one exciter coil, and to avoid filter condensers a closed circuit was used. With the reed stationary, the rest contact of the resonance relay stayed closed and energised the working

relay. But when the reed was vibrating the working relay was released because of its high impedance to the a. c. produced.

The mechanical design of the receiver is shown in Fig. 7. The receiver relay unit and rectifier were mounted in a cylinder of 1 m length and 8 cm in diameter. The cylindrical receiver was put into the fuselage at the tail end. Connection between the receiver and the cabling of the frame was obtained by

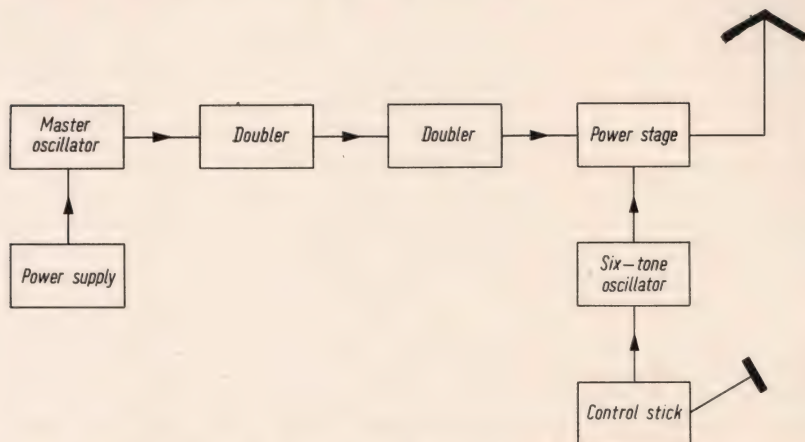


Fig. 8. Radio controlled target simulator. Block diagram of ground transmitter

means of a knife contact strip at the front end. From right to left the receiver contents were: RF-input circuit, detector, regeneration oscillator, first and second LF-amplifier stage, relay unit, measuring jack, and the power rectifier. The equipment was shock-proof mounted on rubber buffers and could be easily changed. The weight was about 3 kg.

Before starting, a ground check was carried out, using an extremal power supply and a plug-in monitor instrument for the voltages and signal levels. The engine was turned over by means of a starter. The aircraft was not launched until the operation of the controls and the performance of the engine had been carefully checked.

#### 4. GROUND STATION

Normally the model was controlled from the ground, and a block diagram of the ground station is shown in Fig. 8. It consisted of a crystal-controlled, amplitude modulated transmitter with frequency quadrupling, a modulator with a six-tone oscillator, power supply, aerial and control stick. The 40 W transmitter was similar in design to the (FuG 203) "KEHL" transmitter, so that it was possible to control the target from an aeroplane.

The control stick was connected to the transmitting vehicle by about 100 metres of cable. Movement of the stick modulated, through contacts, the output of the continuous wave transmitter at the frequencies corresponding to the orders "left-right-up-down". The "ignition-off" signal was given by a button on the stick.

In the final stages of development even untrained people could control the model with surprising accuracy. A series of 100 models was ordered in November 1939.

## 5. RESULTS OF THE FIELD TESTS AND PLANS FOR FURTHER DEVELOPMENT

The very extensive tests carried out during 1939/40 revealed many difficulties which could be eliminated by development. As mentioned above, it was planned to change over from the black-white system to the pulse-ratio method of control.

The use of audio-frequencies and resonance relays for transmitting orders gave trouble because of the fine tolerances necessary on the relays, and the difficulty of obtaining stable audio-frequency oscillators. The simple L-C ground station oscillators had to be tuned rather often with tuning forks, but the effects of this instability could be avoided by using relatively wide audio frequency oscillators in place of the resonance relays.

Great skill was required to land the model. It had to be brought very close to the ground, because the "ignition-off" signal cut off the receiver power supply and the model had to glide in, without controls, and very often crashed. This difficulty was later overcome by the use of a new signal, "choking", which made it possible to bring the model close to the ground with the engine idle.

A comparison of our solution to the light anti-aircraft target problem with the American Rc 56 and 57, made public after the war, shows that the radio control systems were similar.

The receiver Rc 57 described by ACKERMANN and RAPPAPORT in "Electronics" in 1946 had 9 valves, and could deal with five audio-frequency signals, between 650 and 3000 c/sec. It also used super-regeneration and tuned audio frequency circuits for filtering the signal frequencies. A special advantage of the design was the use of a parachute for landing the target, which avoided the tricky radio-controlled procedure.

A grateful acknowledgement is expressed to Dr. F. GOSSLAU and Ing. LAHSE for the photographs and technical information which they made available.

## DISCUSSION

Admiral FAHRNEY (Philadelphia): Under the Bureau of Aeronautics I had the assignment as the project officer for development of target drone aircraft and finally of assault drone or guided missile work. All this radio controlled work was in the full scale aircraft category. Since, Dr. KLOEPFER, all of your lecture concerned only the radio control of model aircraft I would like to know if there was any project in Germany, at the time, to control full scale aircraft by radio?

Dr. KLOEPFER: Radio control of full scale aircraft and missiles was planned from the beginning. The work on the model was done as preliminary study for that purpose. The project was however abandoned since, for the short distance to be covered by the V-1, compass control was sufficient and it was feared that a radio-control system would be jammed.

# GUIDED MISSILES RADIO REMOTE CONTROL

JOSEF DANTSCHER \*

## 1. INTRODUCTION

The aerodynamics, control and tactical purpose of the German guided missiles of the last war have been treated separately in previous papers. In this, and a later report the development of the remote-control systems and the testing of the weapons is described.

Before beginning with this subject, the writer wishes to make a personal statement: In writing this historical report on a far advanced technical development, we are faced with a strange situation: we have to write this report almost from memory, without access to the reference material. The reports which we wrote in the past are still in existence, but are not available to their writers. In April of 1945 we surmised that some day we might be faced by such a task. We prepared for this situation but fate had it differently.

For this reason I must ask for the indulgence of those experts who took up the threads of our development work, which we began during the war, and which they continued thereafter. Our peculiar situation touches a general problem of engineering in our time. Many engineers became aloof from their former creations and considered them as something with which they were no longer concerned, and whose exploitation they left to others, perhaps only too gladly. However, gradually one begins to grasp that the originator still carries a responsibility for his creation. Whether he likes it or not, he must care for the use of his creation. He must continue the study of its conditions, the limits of its application. For only the originator has the full judgment and understanding of his own creation. He cannot afford to shirk this responsibility, but must, if necessary, claim it and carry it through. This applies also to the subject of this meeting. For it deals with a highly developed technical device for combat use.

Permit me touch another point of view. This paper deals with work done a long time ago. In the meantime the originators had to refrain from their work, while it was continued by others. They did so, partly because of restrictions, partly of their own will. The present state of the art is, therefore, far advanced compared with what it was in their own time. Perhaps, however, this will permit them to see things from a distance, and will aid them in their historical task.

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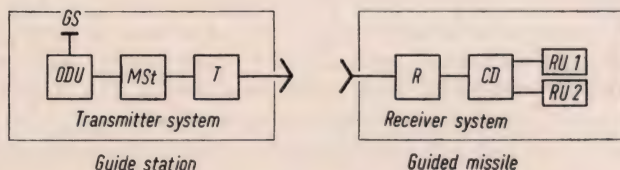
## 2. PRINCIPLE OF THE RADIO REMOTE-CONTROL SYSTEM

The tactical purpose of the remote-control system is to transmit to the bomb steering orders, after its release from the aircraft, by which its flight path can be corrected. This is to correct inaccuracies of release, drift, errors in measuring the altitude and evasions of the moving target.

In the papers dealing with the fundamental problems of the guided missiles, it has been stated that the method of visually covering the target by the missile was used to correct its flight path. This method requires control orders to incite transverse movements to the flight path, depending on the controllability of the missile.

The system of co-ordinates that is used may be either the Cartesian system, involving fore and aft, left and right orders, or a polar system of co-ordinates, involving push or pull, left roll or right roll orders. The transmitting system thus has the task of transmitting continuously, throughout the time of flight of the bomb, two independent, qualified amounts of control.

The basic scheme of the KEHL/STRASSBURG control system is shown in Fig. 1.



*Fig. 1. Basic scheme of radio remote control*

*GS = Guide stick, ODU = Order distributor unit, MSt = Modulator stage, T = Transmitter, R = Receiver, CD = Connecting device, RU = Rudder actuating sets.*

By means of a guide stick (GS) controlling a modulator stage (MS), the given orders modulated the output of a transmitter (T) in the guiding aircraft. The H.F. signals were de-modulated by the receiver (R) in the missile and the orders were passed on to the two sets of steering means through a connecting device (CD). The missile thus could be controlled about both axes corresponding to the movement of the guide stick. Owing to the short time of flight — about a minute — only a continued guiding could be used. The amount of the guiding order was determined by the time-ratio  $T_1$  and  $T_2$  of two alternating and periodical modulation frequencies  $f_1$  and  $f_2$ . The use of two modulation frequencies for each order reduced the susceptibility to jamming as compared with the use of a single audio frequency. The order  $K$  thus amounts according to Fig. 2 to

$$K = \frac{T_1 - T_2}{T_1 + T_2} = \frac{T_1 - T_2}{T},$$

where  $T_1$  is the duration of the modulation of the frequency  $f_1$  and  $T_2$  is the duration of the modulation of the frequency  $f_2$ .

The alternating switching of the modulation frequencies was done by means of a guide stick which served as an order distributor. The various ratios of the two frequencies switched were produced by spiral segments on rotating cylinders in the order distributor unit. Both co-ordinates of the co-ordinate

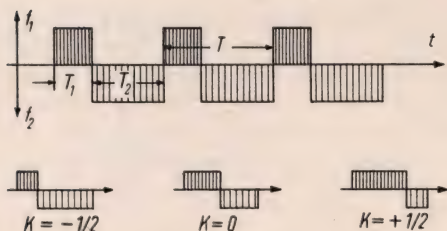


Fig. 2. Order amount in the case of two-frequency modulation

$f_1, f_2$  = Frequencies,  $t, T, T_1, T_2$  = Time,

$$K = \frac{T_1 - T_2}{T_1 + T_2} = \frac{T_1 - T_2}{T}$$

system were served by one switching cylinder each. The desired amount of orders was picked up along the axis of the cylinders, in accordance with the location of the guide stick.

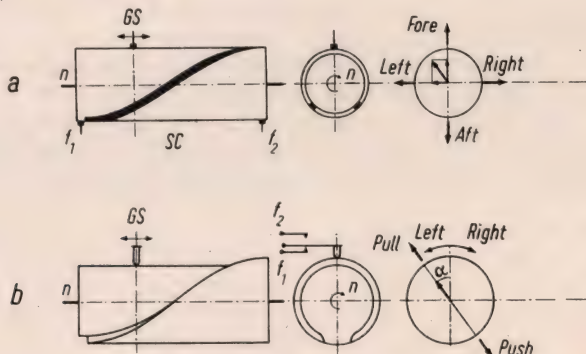


Fig. 3. Order distributor unit

a) Kehl I. System of Cartesian co-ordinates.

b) Kehl III. System of polar co-ordinates.

GS = Guide stick, SC = Switching cylinder,  
 $f_1, f_2$  = Frequencies,  $n$  = Revolutions per minute.

Fig. 3 shows schematically the arrangement of switch cylinders of both systems. In the case of FRITZ X the distance of the guide stick from the centre in the two directions of the Cartesian system of co-ordinates was transmitted as order amount, but in the case of Hs 293 the order amount was formed by the angle of turning the guide stick about its axis, designating the desired location of the transverse axis of the missile, and the inclination of the guide stick at a right angle to this — polar co-ordinates.

In the case of the KEHL I radio control system for FRITZ X the switching was done directly at a frequency of 5 c/sec by driving the switch cylinders at 300 r.p.m by means of a speed regulated D.C. motor. In the KEHL III system for the Hs 293 glider bomb, the speed was 600 r.p.m. and the switching frequency 10 c/sec.

In a later development, the mechanical switching of the modulation frequency was effected by means of a multivibrator, known as flip-flop guide stick "KLAPPER" with a "KARTE"- or "POL" device for Cartesian and polar coordinate control, respectively. This system was destined for use with the "KOGGE" radio control system, operating on the decimetre band. This improvement considerably reduced components amount, insofar as it had to be accessible to the bomb-aimer.

### 3. THE KEHL/STRASSBURG RADIO CONTROL SYSTEM FOR THE FRITZ X FREE FALLING BOMB

Fig. 4 shows a schematic diagram of the "KEHL" (FuG 203) transmitting system. The frequencies of modulation were produced in the modulation unit, and were switched by the order distributor unit. The modulation was done at frequencies of 1.0, 1.5, 8 and 12 kc/sec.

The crystal-controlled transmitter worked on 18 channels, spaced 100 kc/sec apart, between 48 and 50 Mc/sec, so that it was possible to launch up to 18 bombs simultaneously.

The crystals were tuned to half frequency and drove the transmitter through a doubler. The output stage had two LS 50 valves of about 30 Watt modulated output power (high frequency power 70 Watt) at 40% amplitude modulation. The power was fed to the antenna through a matching device. The set as a whole corresponded to the customary conditions for aircraft use.

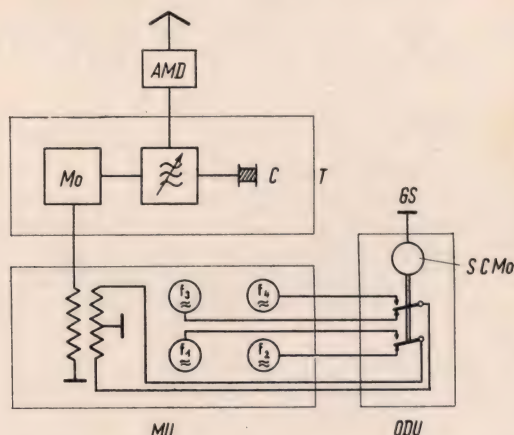


Fig. 4. Basic scheme of Kehl transmitting system

AMD = Antenna matching device, T = Transmitter, MU = Modulation unit, ODU = Order distributor unit, Mo = Modulator, C = Crystal, GS = Guide stick, SCMo = Switching cylinder motor,  $f_1, \dots, f_4$  = Frequencies.

The receiver set (FuG 230) (Fig. 5) for the FRITZ X free falling bomb consisted of the "STRASSBURG" receiver, which was connected to the antenna by means of a matching device. The output relay of the receiver set actuated the magnets of the spoiler switching system directly. In the case of the FRITZ X

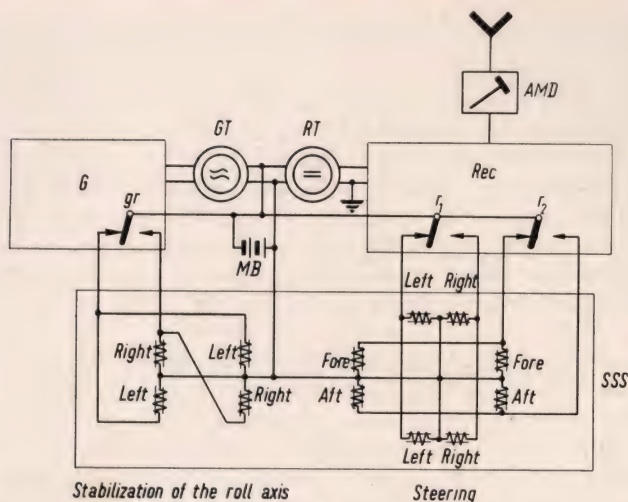


Fig. 5. Basic scheme of Fritz X radio receiving system (Fu G 230 a receiving system)  
 AMD = Antenna matching device, G = Gyroscope set, GT = Gyroscope generator, RT = Receiver generator, MB = Missile battery, SSS = Spoiler switching system, gr = Gyroscope relay,  $r_1, r_2$  = Relay contacts.

free falling bomb, in order to keep the co-ordination between control order and spoiler steering effect of fore-aft and left-right, the rotation of FRITZ X about its roll axis had to be prevented by a gyroscope and separate spoilers.

On long flights, it was possible for the temperature of the bombs to fall considerably, and since this would have influenced the internal humidity, the bombs were heated by controllable warm air supplied through a hose from the aircraft. This was especially important in view of the capacity of the battery within the bomb. The temperature was supervised by means of a temperature pick-up system in the bomb and an indicator in the aircraft.

The "STRASSBURG" receiver set worked on the heterodyne system and by means of a screw driver could be connected to any one of the 18 channels. The variations were compensated by an automatic high-accuracy tuning device

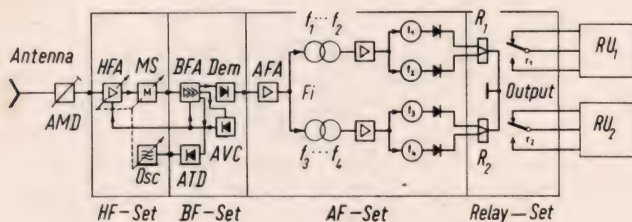


Fig. 6. Basic scheme of Strassburg receiving system

AMD = Antenna matching device, HFA = H.F. amplifier, MS = Mixer stage, Osc = Oscillator, BFA = Beat frequency amplifier, Dem = Demodulator, AVC = Automatic volume control, ATD = Automatic tuning device, AFA = Audio frequency amplifier, Fi = Filter,  $f_1, \dots, f_4$  = Frequencies,  $R_1, R_2$  = Relays,  $r_1, r_2$  = Relay contacts, RU = Rudder actuating sets.

( $\pm 35$  kc/sec) together with good temperature compensation. A schematic diagram is shown in Fig. 6. Large variations of the receiving potential, caused by the growing distance from the launching point and by interference, made it necessary to use an automatic volume control. For a satisfactory sensitivity of about  $2\mu\text{V}$ , a regulation of  $1:10^5$  was possible. The original type (E 30) had two RL 12 P 10 S switching valves in the output, but on the production model (E 230) two relays (T.rls 64) were used. Before the FRITZ X was launched the "KEHL" transmitting system and the "STRASSBURG" receiving system were connected by a 14-pin "break-off" connector. Its purpose was to prepare the start of, and to start, the transmitting and receiving units in proper time, to actuate all functions in the proper order, and to control the various consecutive switching conditions.

The connection diagram is shown in Fig. 7. The receiver also was connected, via a main switch, to the main battery of the aeroplane. When the system operating switch was put in position 1, the various sets were heated, and the gyroscope generator and the gyroscope started rotating. After one minute the gyroscope rotated at full speed. When the switch was put in position 2, the transmitter and receiver generators and the order distributor switching cylinders began to rotate. The bomb was then ready for launching.

The current meters provided a means of checking that the transmitter and receiver were working correctly, and by means of the selector switch (*SeS*) any

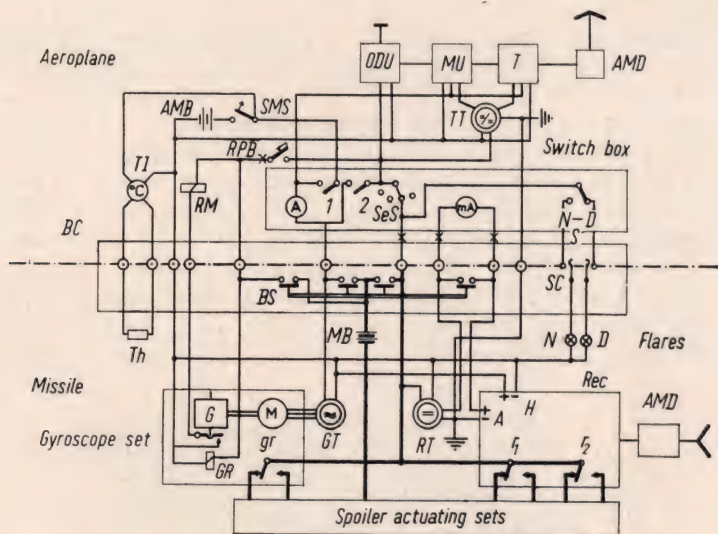


Fig. 7. Basic scheme of Fritz X radio control Kehl/Strassburg system

ODU = Order distribution unit, MU = Modulation unit, T = Transmitter, AMD = Antenna matching device, AMB = Aeroplane main battery, SMS = System main switch, TT = Transmitter transformer, RPB = Release push button, RM = Release magnet, BC = Break-off connector, TI = Temperature indicator, Th = Thermoelement, SeS = Selector switch, A, mA = Current indicator, N-D S = Night-day switch, SC = Sliding contact, BS = Break-off switch, MB = Missile battery, G = Gyroscope, M = Gyroscope Motor, GT = Gyroscope generator, GR = Gyroscope release, gr = Gyroscope relay, A = Anode, H = Heating,  $r_1, r_2$  = Relay contacts, Rec = Receiver, RT = Receiver generator.

particular bomb could be selected for launching. It was possible to selectively launch up to four bombs by one unit.

By pressing the release push button *RPB* or via the release contact of the bomb sight the gyro in the bomb was released and at the same time closed the gyroscope contact *RM* which energised the magnet in the releasing mechanism.

When the break-off connector *BC* separated and the contacts *SC* passed one another, the previously switched day or night flare charge was ignited. The break-off connector switch *BS* disconnected the heaters and generators of the bomb from the main battery of the aeroplane, and connected them to the bomb battery. The receiver set then worked independently and the bomb was ready for the remote control.

The diagram is considerably simplified. For example, the selector switch and the system operating switch were mutually locked and the launching was not actuated directly by the push button, but via a relay, which could only be actuated with the gyroscope still settled.

In order to limit the high frequency radiation only to the flight time of the bomb, the transmitter was not switched on until the instant of release. During the first seconds after the release, a time switch, set in motion by the contact of *GR*, greatly reduced the radiated energy by means of a resistance within the antenna matching device. The purpose was to avoid over-control of the receiver while the bomb was still at a short distance.

#### 4. THE "KEHL/STRASSBURG" RADIO-CONTROL SYSTEM FOR THE Hs 293 GLIDER BOMB

In principle and in fact, the transmitting system for the remote control of the Hs 293 was the same as that for the FRITZ X, even though in the early stages of development it had been constructed as a separate system ("KEHL III"). Consequently the "KEHL IV" system was developed to serve both the FRITZ X and the Hs 293. It was installed in the large aeroplanes DORNIER Do 217, FOCKE-WULF Fw 200 and HEINKEL He 177. Since they were practically alike, they will not be discussed further.

The receiver set of the Hs 293 was more extensive, because the control mechanism was not separated from the stabilizing mechanism. While the FRITZ X used a direct connection to the spoiler sets, its stabilization being entirely independent of the remote control system, in the case of the Hs 293 it was necessary because of the particular kind of control mechanism and its connection with the stabilizing mechanism, to use a special connection device to co-operate with the existing stabilization elements.

This is schematically explained in Fig. 8 by a comparison of the two principles of the direct and indirect connection. In the case of direct connection, the receiver operated the rudders directly, but in the case of indirect connection, the output of the receiver actuated servomotors or similar devices to operate the rudders. Once the desired positions of the rudders had been reached, the servos were stopped by a feedback command. These two functions were done by a special connecting device between the receiver output and the rudder operating means.

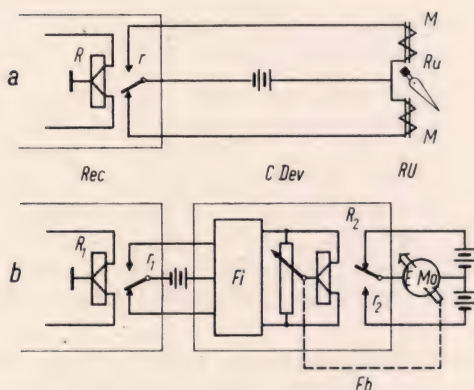


Fig. 8. Basic scheme of the connecting device

a) Direct connecting b) Indirect connecting with feedback

Rec = Receiver, C Dev = Connecting device, RU = Rudder actuating set,  $R, R_1, R_2$  = Relays,  $r, r_1, r_2$  = Relay contacts,  $M$  = Magnet,  $R_u$  = Rudder,  $F_i$  = Filter,  $E Mo$  = Elevator motor,  $F_b$  = Feedback.

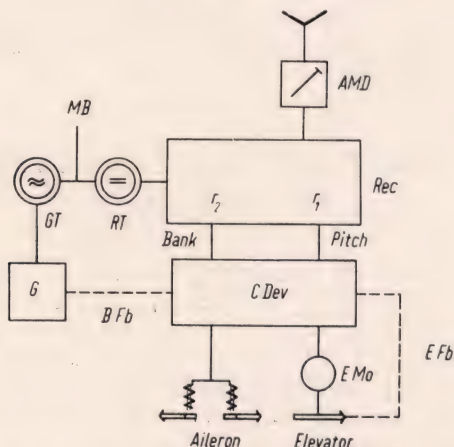


Fig. 9. Basic scheme of the Hs 293 radio receiving system (Fu G 203 b receiving system)  
 AMD = Antenna matching device,  $G$  = Gyroscope,  $GT$  = Gyroscope generator,  $RT$  = Receiver generator,  $MB$  = Missile battery,  $Rec$  = Receiver,  $r_1, r_2$  = Relay contacts,  $C Dev$  = Connecting device,  $E Fb$  = Elevator feedback,  $B Fb$  = Bank feedback,  $E Mo$  = Elevator motor.

A schematic diagram of the arrangement and the connection of the units in the case of Hs 293 is shown in Fig. 9. The elevator motor  $E Mo$  was started, by the connecting device, until the feedback indicated that the desired position had been reached. The ailerons were actuated until, under the control of the gyroscope  $G$ , the required lateral inclination was reached. In this position, however, the ailerons were not at rest but flapped up and down a little with the period of the switching frequency. This helped to speed up their response.

Details of the interconnection between the connecting device and the rudder sets are shown schematically in Fig. 10.

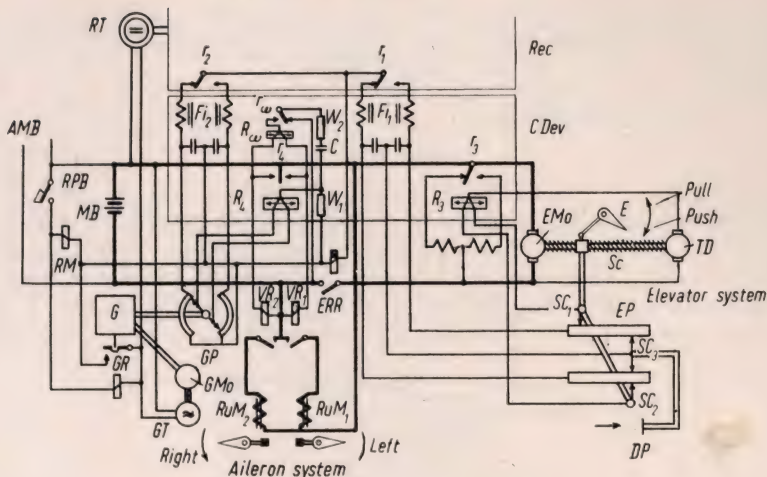


Fig. 10. Hs 293 connecting device

RT = Receiver generator, AMB = Aeroplane main battery, RPB = Release push button, MB = Missile battery, RM = Release magnet, G = Gyroscope, GR = Gyroscope release, GP = Gyroscope potentiometer, GT = Gyroscope generator, GMo = Gyroscope motor, RuM = Rudder magnet, Rec = Receiver, CDev = Connecting device, E = Elevator, EMo = Elevator motor, ERR = Elevator retarding relay, EP = Elevator potentiometer, TD = Tacho dynamo, SC<sub>1,2,3</sub> = Sliding contacts, DP = Drag plate, R = Relays, r = Relay contacts, C = Capacity, W = Resistances, Fi = Filters, VR = Vacuum relays.

The elevator order excited the field windings of a motor via  $r_1$  and  $r_3$  for left or right hand rotation, which moved the elevator by means of a screw. The necessary feedback was produced by the potentiometer  $EP$ , the two windings of which were driven symmetrically by the elevator screw. The governing was aided and oscillations avoided by means of a damper, consisting of a tachodynamo  $TD$ , connected to the elevator motor screw, which produced a voltage proportional to the speed of rotation. Since the required elevator angle depended on the airspeed, the elevator potentiometer was influenced by the drag of the plate  $DP$ .

The aileron-order from the receiver was fed to the gyroscope potentiometer  $GP$  via a filter  $F_2$ . The pick-up was controlled by the gyroscope, and actuated the relay  $R_4$  (and its contacts  $r_4$ ) and further vacuum relays  $VR_1$  and  $VR_2$  and thus operated the ailerons. In this case also any oscillations were damped by a voltage proportional to the roll speed which was fed to the resistance  $W_1$ , by means of the relay  $R_6$ .

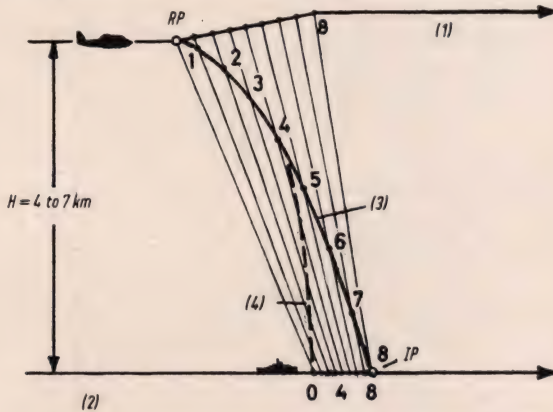
The inter-connection with the transmitting set before release and the functions of the "break-off" connector were the same as on FRITZ X. Here also it was necessary to release the gyroscope and, by means of the gyroscope release contact *GR*, to excite the release magnet, before the bomb could be released by pressing the push button. I shall not, therefore, explain this inter-connection.

However, it is noteworthy that the elevator machine could not be actuated until one second after the release. The delay was obtained by the thermic

retardation relay *ERR*. This was to prevent a collision between the released bomb and its aeroplane until the Hs 293 had gone ahead under its own propulsion.

## 5. ANTENNAE

Before explaining the conditions of the antennae, the flight paths of *FRITZ X* and Hs 293 will be recalled. Fig. 11 shows the launching of *FRITZ X*. It can be seen that only a very small spacial angle may be used for transmission and reception. In the same way the launching of the Hs 293 required a very small spacial angle, as is shown in horizontal and vertical projection in Fig. 12. In



*Fig. 11. Fritz X free falling bomb remote control*

- |  |                           |
|--|---------------------------|
| (1) = Aircraft flight path,            | (2) = Target course,      |
| (3) = Flight path of the guided bomb,  | (4) = Trace of free fall. |
| RP = Release point, IP = Impact point. |                           |

this case, however, the necessary turn of the aeroplane required a much larger transmitting antenna angle. The system of visually covering the target by the bomb reduced the spacial angle which the bomb required for its reception from the rear to a very small amount. In both cases it was necessary, to have the correct antennae diagram within this spacial angle, and this was guaranteed by correct arrangement of both the transmitting and receiving antennae. Furthermore, these conditions made it desirable to have the sensitivity of the antennae in all other directions as small as possible. The purpose was to make it as difficult as possible for the enemy to detect remote control and the frequency used in it, and also to reduce the possibility of jamming.

These considerations induced us to plan as a further development of the "KEHL/STRASSBURG" system, in the ultrashort wave range, the "KOGGE" device within the decimetre band. Using this ultra high frequency it was possible to develop antennae having the desired directional characteristics.

The transmitting antennae were wires extending from the centre part of the fuselage of the aeroplane to the tips of the tailplane; they were connected to the transmitter by means of the antenna-matching device and a  $60 \Omega$  high frequency

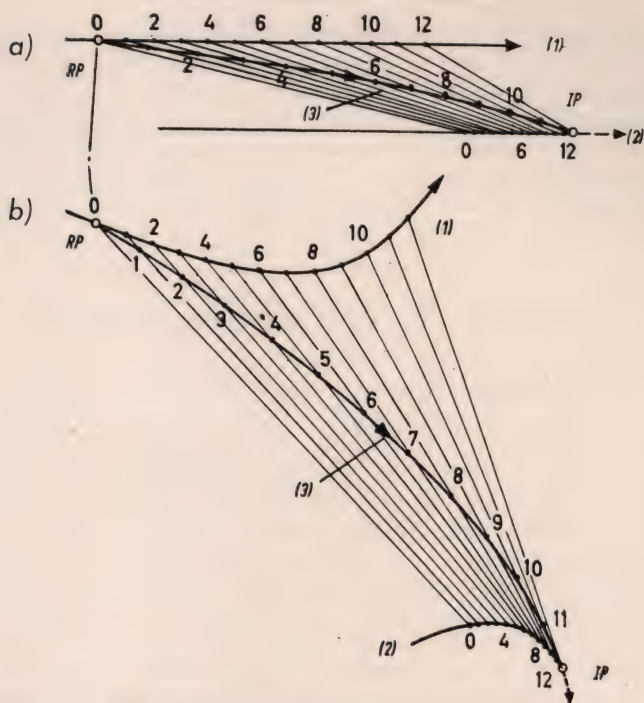


Fig. 12. Hs 293 glider bomb remote control

a) Vertical projection. b) Horizontal projection.

(1) = Aircraft flight path,

(2) = Target course,

(3) = Flight path of the guided glider bomb.

RP = Release point, IP = Impact point.

cable. The characteristics of these antennae produced good radiation within the required spacial angle. In the case of the FRITZ X free falling bomb one of the air-brake tubes insulated within the ring surrounding the tail surfaces was used as an antenna (Fig. 13). The antenna-matching device was mounted within

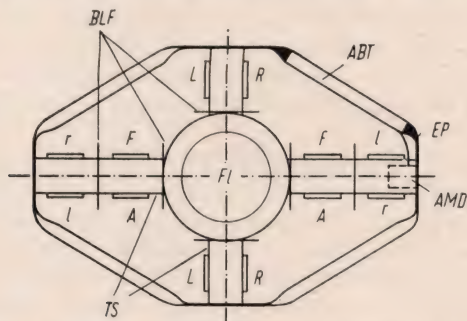


Fig. 13. Antenna arrangement in Fritz X tail plane

BLF = Boundary layer fences, ABT = Insulated air brake tube serving as antenna, EP = End plate, AMD = Antenna matching device, TS = Tailplane shells, Fl = Flare,

L = left, R = right, F = Fore, A = Aft, l, r = Roll stabilization.

the shell of the tailplane near the outer end plate. From there, a 60  $\Omega$  high frequency cable led to the receiver input. The reception from the rear was very good.

In the case of the Hs 293 glider bomb the antenna led from the middle of the right side of the fuselage to the right elevator tip, and from there above the elevator flap across to the left tip. The antenna-matching device was mounted at the plexiglass-pane of the lead-in and could be tuned from the outside.

## 6. JAMMING OF THE RADIO TRANSMISSION

Right from the beginning great attention had been paid to the possibility of jamming the radio transmission and this problem was already decisive when the "KEHL/STRASSBURG" radio-control system was planned. A big advantage of the ultrashort wave was that its quasi-optical propagation made observation and jamming by the enemy impossible, if the distance was large and the altitude small. It was made even more difficult by the short duration of the transmission. Furthermore, the likelihood of jamming was reduced by efficient high frequency selection, the use of 18 channels, and the possibility of changing the frequency in the case of an observed jamming. Also the audio-frequency selection of four discrete modulation frequencies (two frequencies being used for the transmission of each command) together with the use of switching frequency made it improbable that an attempt to jam by giving false orders would be successful.

During the tuning of the aircraft system sets on the ground before take-off, the time and energy of transmission were limited to a very minimum. A further means of rendering observation and jamming more difficult was the directional characteristic of the antennae (see above).

It was necessary to take into account the eventual case of a remote control aircraft being shot down, and the devices thus falling into the hands of the enemy. In order to make it as difficult as possible to obtain information on the remote control method from such captured equipment, all essential units were provided with centrally released explosive destruction charges and burstable unit construction elements. Therefore in the case of an aircraft being shot down, one could rely on a destruction of all essential parts of the "KEHL" remote control installation. Most of the operations were carried out over the sea and this was another factor which reduced the chance of an installation falling into the hands of the enemy.

Two more bands of 60 and 27 Mc/sec were provided as alternative devices with 9 frequencies each, so that in the case of a detection by the enemy of the frequencies used, the operations might be continued. These bands were available from supply points for instantaneous re-equipment of the combat formations. Furthermore, as already stated, another installation (KOGGE) using decimetre waves, but corresponding in all other details to the "KEHL/STRASSBURG" system, was being developed. The modulation frequencies were 6, 9, 13 and 16 kc/sec. This device was also to be used as an alternative equipment against eventual jamming. It made full use of the directional characteristics of the antennae and was remarkably safe against detection as well as jamming. On the other hand

the development did not go beyond a few sample devices or pilot series, and it did not go into production.

Another method of preventing jamming by the enemy might be mentioned, i. e. the development of remote control by a line guidance system, but this is dealt with in another paper.

In order to investigate the possibility of jamming the "KEHL/STRASSBURG" remote control system a number of investigations were carried out by the Commissioner of High Frequency Research. The possibility of jamming by continuous dashes, impulse, wobbling and feedback transmitters was thoroughly investigated. This showed that, in spite of the system of double selection achieved by the use of audio frequency modulation, it was possible to prevent desired control, by means of an amplitude modulated jamming transmitter, which was wobbling within the range in question. It was possible to develop the required power of transmission in the zone of operations. According to these investigations, the reason for this may be that audio frequencies are produced by some non-linear connecting element, e.g. the first demodulator of the receiver, by a superposition of the control wave and the jamming wave. In the case of a wobbling jamming, the audio frequencies provided for the order transmission are, interrupted by the wobbling frequency, only short and produce faulty orders. A high wobbling frequency reduces the jamming effect by the shorter duration of the dangerous frequencies, but the possibility of this type of jamming had to be taken into account. In a further study by the same office, a mock transmitter for the launching and tuning of Hs 293 and FRITZ X was developed. Launching- and tuning transmissions were simulated by special control programs with frequencies of modulation which were different from the real frequencies, and with four side bands on each side of the radio transmission frequency. This was to make it difficult for the observation service to detect the frequency of radio transmission actually used.

In order to study the possibility of the use, in combat, of the "KEHL/STRASSBURG" system, and to determine the alternative wave ranges and to map the areas of jamming as determined by the enemy's radar and navigational radio beacons, a trip was made into the coastal areas of Holland, Belgium and Northern France with a "STRASSBURG" receiver. The result was that in the normal range of frequencies, the frequencies K 1 to K 9 were jammed by the hyperbolic navigation transmitters. Within the coastal area between Dünkirchen and Berk, and over all southern and southeastern England, and coastal waters including the Isle of Wight, this frequency range was not advisable for combat use. The range of frequencies K 10 to K 18 was free from jamming in the channel and the adjoining coastal areas. Only in the close neighbourhood of the transmitters of the East-System of hyperbolic navigation ( $f = 48.1 \text{ Mc/sec}$ ) would a launching have been impossible.

There was no jamming observed in the range of alternative frequency 1, because in these frequencies the enemy had no radar or other powerful impulse transmitters. A launching with these frequencies would have been unjammed in the coastal area and over the English main-land.

The alternative band 2 was in the range of frequencies of the powerful British radar sets, and jamming could be expected in the same coastal range around Cap Griz Nez and along the English coast. The use of this alternative equipment was, therefore, not advisable in that area.

In a further study, based on the investigation of the possibility of jamming, a proposal was worked out in which two separate channels were used, in both the transmitter and the receiver. By additional circuit elements the audio frequencies in both channels were transmitted with the opposite phase. This made it possible to separate the order impulses from the jamming impulses.

Another investigation by a University Institute was concerned with the disturbance of the radio transmission by the jet stream of the reaction drive of the Hs 293. It was shown that a disturbance is produced by the existence of dispersed particles in the zone of the jet stream, but that for the Hs 293 this type of disturbance was unimportant. The investigation showed that, by proper methods, or by the combination of various methods, the charging effects, causing the disturbance, can be practically eliminated. Possible methods are the injection of water into the narrowest nozzle-area, the lining of the inside of the nozzle by so called cold putties (method of lacquer strips) and by the application of discharge sticks at the end of the nozzle (muzzle method). It was shown that the charging effects could be completely overcome by the use of these methods, but for the reasons given above they were not used practically.

## 7. THE TESTING OF THE REMOTE CONTROLLED MISSILES

A few general points might be mentioned which refer to the testing of these novel devices. It should be recognized that in a technical development for combat use where everything was so completely new, a number of general questions and relations were of the greatest importance. Treating and solving of the special problems therefore, was only possible on the base of synopsis and consideration of those general circumstances. It is characteristic of new technical weapons that the development cannot start from such detail things as the missile structure, remote control system, combat operation or tactical use. It is necessary to begin with the technical possibilities, and a proper solution must be found by using these possibilities to the full and by being acquainted with all detail problems. However the reverse influence of the details upon the whole must be considered. In this way only can a weapon be created that will bring essential advantages, is capable of being improved, and can be used successfully in combat.

It is very difficult for industry or any military organization to meet these conditions, and for this reason the testing is most important. The testing organization must compose the partial solutions contributed by the industry as well as by the operators. Before the new devices are adopted, the testing organization must test them critically and express an opinion. The testing organization must also digest the experience obtained from the use by military units and in combat. Such a synopsis is beyond the means of industry and the operating forces. The testing is made the more difficult because at the end of each testing the object to be tested is destroyed and lost and an inspection result is scarcely available. It requires very well prepared tests and exhaustive measuring methods to determine their performance, taking into account the failures and the faults. Versatile and profound competent experience is required too in order to be able to evaluate the results.

The technical testing of the devices was carried out with the co-operation of industry by the test base of Peenemünde and the introduction to military use was done by testing commands of the forces. Between the two there was close co-operation and a vivid exchange of experience.

The Hs 293 was designed and built in 1939/40. The testing was taken up in the autumn, and on December 18<sup>th</sup> the first successful launching test was carried out. Thereafter a few changes of design had to be carried through, such as the removal of a stop on the gyroscope, which caused a precession movement, and the application of the propulsive system. The important influence of humidity upon the functioning and storing of the bomb was recognized. The questions of the antennae and the transmission had to be cleared; testing devices had to be developed; the method of checking in operational use had to be determined. The following testing devices were required: A testing transmitter, testing voltmeter, testing receiver, testing battery box, a tester for break-off connector, testing diagrams, instruction devices. This includes the remote control instruction device, which is not dealt with in this paper. By the end of 1941 the Hs 293 was declared ready for combat.

In spite of the greater simplicity of the FRITZ X, its development took longer than that of the Hs 293, and it did not become ready for combat until the autumn of 1942. During the introduction to military use the delay was recovered, thanks to the experience gathered in the meantime by the military testing group. Thus the FRITZ X was operational at the same time as the Hs 293.

## 8. COMBAT EXPERIENCE

A short word on the combat experience of both bombs, insofar as it refers to technical questions. During the training of the combat-squadron KG 100 various difficulties and failures occurred. As a rule, they were caused by insufficient, or negligent, checking by the operator, or by improper use. The same was observed in combat. There did not appear any fundamental technical faults. An important difficulty was the lack of technically minded persons in the higher commands and the lack of understanding there of the technique of this special weapon and its requirements, and the temporary unreliability of the checking done by the operators. It was not the device that proved to be faulty, but the men who used the device. Finally an industrial service and repair organization was formed by the TELEFUNKEN Company, which supported technically the combat units and supply points and equipped the aircraft at producer plants.

The above preconditions for the successful use of remote controlled weapons may have been the reason why the Western powers became interested in their development results and importance only after the Eastern powers had continued, with great zeal, the development which had fallen into their hands. The combat experience of both FRITZ X and Hs 293 has proved the usefulness of this weapon. About 100 bombs were launched against the enemy within one year, beginning in the summer of 1943, in the Mediterranean, the Atlantic and the North Sea. The operations were not always started with good understanding, yet two thirds of the bombs reached the targets. One half of these bombs were

full hits and effective near hits, respectively. This could not have been obtained with conventional bombs. It proves the practicability of remote control.

The possibilities offered by this development have not been fully utilized. The reason was the lack of technical understanding by the higher commands. They did not understand the whole complex of questions involved in combat with such special technical weapons, causing a tactical uncertainty by this. They hesitated incredibly before the first operation. In the meantime the enemy gained the air superiority. But even then the inference of the change of conditions was not drawn. The possibilities of these weapons when used against certain land targets were realised too late.

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# FURTHER DEVELOPMENT OF REMOTE CONTROL SYSTEMS AND THE REMOTE CONTROL OF AIR-TO-AIR MISSILES

JOSEF DANTSCHER \*

## 1. INTRODUCTION

In order to prevent jamming, a line guidance system with two-wire transmission was developed, using audio frequency for the Hs 293 glider bomb and direct current for the FRITZ X free falling bomb. The Hs 298 air-to-air missile was guided with a radio control system of the "KEHL-COLMAR" type, the X-4 with a transmission line system of the "DÜSSELDORF-DETMOLD" type. The latter represents the simplest kind of remote control system. A summary of all the systems developed for guiding airborne bombs is given at the end.

## 2. AUDIO FREQUENCY LINE GUIDANCE SYSTEM

The radio control of the bombs FRITZ X and Hs 293 has been described in a previous paper. The possibility that the system might be jammed was considered, and suitable preventive measures were discussed. In order to ensure successful delivery of the missile, even in the event of effective enemy jamming, the "KEHL-STRASSBURG" control system was adapted for use with a two-wire transmission line. Two methods were adapted for this purpose, one with audio frequency, the other with direct current, transmission.

The audio-frequency two-wire system was known as "DORTMUND-DUISBURG" ("DORTMUND"-transmitter, "DUISBURG"-receiver), or as FuG 207/FuG 237; it is shown schematically in Fig. 1. Similar to the radio control system the audio frequency was switched by a guide stick, amplified, and fed into the line through a matching unit. At the far end of the line the signal entered the receiver via another matching unit and passed through a connection device to the rudder sets. Fig. 2 shows the circuits of the frequency generator and the receiver discriminator. One switch element of the order distributor unit applied a positive or negative bias to a diode, so that the two circuits consisting of C and L, or of C and L with L' in parallel, oscillated in turn. In this way the two frequencies  $f_1$  and  $f_2$  were switched. The receiver discriminator was tuned to the mean frequency, so that positive and negative voltages were fed, in turn, into the output relay. The second switch element in the order distributor unit was

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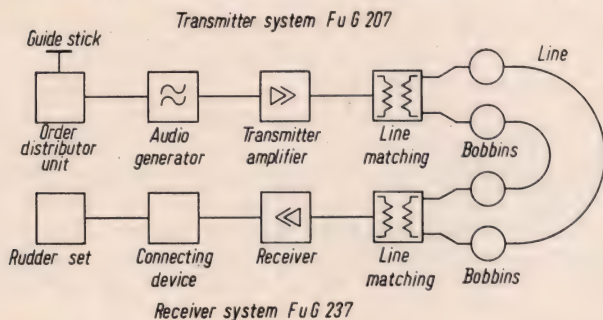


Fig. 1. Basic scheme of the wire-guidance system (A.C.) "Dortmund - Duisburg"

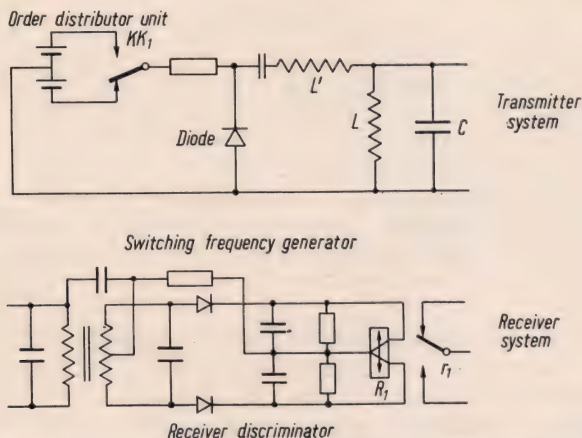


Fig. 2. Basic scheme of the audio frequency-guidance system

linked to a similar circuit arrangement using the same transmission line. The mean frequencies were 450 and 700 c/sec respectively, therefore with  $\pm 5\%$  detuning the four line frequencies were 427, 473, 665 and 735 c/sec.

According to reports in foreign journals, based upon enquiries at the STASSFURTER RUNDfunkGESELLSCHAFT, they are said to have been 450, 550, 650 and 750 c/sec. The choice of frequencies was a compromise between small size of apparatus — a high frequency meant lighter components — and damping on the line — damping is greater at high frequencies. The receiver was fitted with a very effective automatic volume regulation which, due to the changing length of the line, had to compensate for an input voltage variation of the order of 1 to  $10^5$ . The RG 12 D 2 valve was used throughout as diode.

### 3. THE DIRECT CURRENT LINE CONTROL SYSTEM

The direct current transmission system had the code name "DÜREN-DETMOLD" ("DÜREN"-transmitter, "DETMOLD"-receiver) or Fu G 208/Fu G 238. Its principle is shown in Fig. 3. One of the orders was transmitted by switching the direct current polarity, the other by switching the intensity of the current (high/low).

The contacts  $KK_1$  and  $KK_2$  were operated from the guide stick, by means of a relay system with two potentiometers, at a switching frequency of 5 c/sec. The periodic reversal of the direction of the current by means of  $KK_1$  operated the relay  $R_1/r_1$  at the receiver. Contact  $KK_2$  periodically short-circuited the resistance  $W$  and thus caused relay  $R_2/r_2$  in the bomb to operate, but with  $W$  in circuit this relay released. Relay  $R_1/r_1$  was adjusted so that, even with low current ( $W$  in the circuit), it was certain to operate to one side or the other, depending upon the polarity.

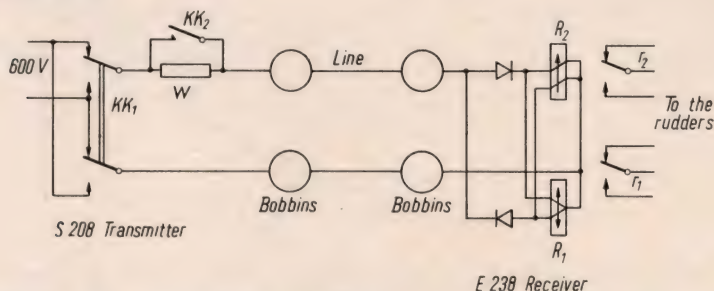


Fig. 3. Basic scheme of the wire-guidance system (D.C.) "Düren-Detmold"

This transmission system was the simplest of those developed during the war, regarding the number of components. The transmitter consisted of three switch elements and one resistance, the receiver of two rectifiers and two T-relays.

Originally it was intended to develop both systems (audio frequency and direct current) for both types of bombs, i.e. for FRITZ X and the Hs 293, but this plan was later abandoned because of the problematic manufacturing- and supply-capacity. The audio frequency system was used exclusively with the glider bomb Hs 293, the direct current system with the free falling bomb FRITZ X. However, the growing reduction in industrial capacity, and the small share of FRITZ X in the total number of operational flights, eventually made it necessary to abandon the production of the FuG 208/FuG 238 altogether, and only the "KEHL-STRASSBURG" radio control system was kept in stock for use with FRITZ X.

#### 4. BOBBIN ARRANGEMENT FOR LINE CONTROL SYSTEMS

The construction of the bobbins, and the smooth running of the wire, presented considerable difficulties. In the initial stages, Dr. KRAMER and Prof. WAGNER tried separately to find solutions for the FRITZ X, and the Hs 293, respectively, but although they made substantial progress, a satisfactory solution was not obtained until they combined their knowledge. It was found that a bobbin of the correct shape was essential to ensure proper functioning at the high running speeds and low outside temperatures.

The bobbins were attached in pairs to the aircraft and to the bomb body. The wire used was steel piano wire of 0.2 mm diameter, wound up as a cylindrical body with a certain amount of pre-torque to avoid kinks when running off. Fig. 4 shows the arrangement for the glider bomb Hs 293. Here the bobbins were fastened to both trailing wing tips in place of the air-brake

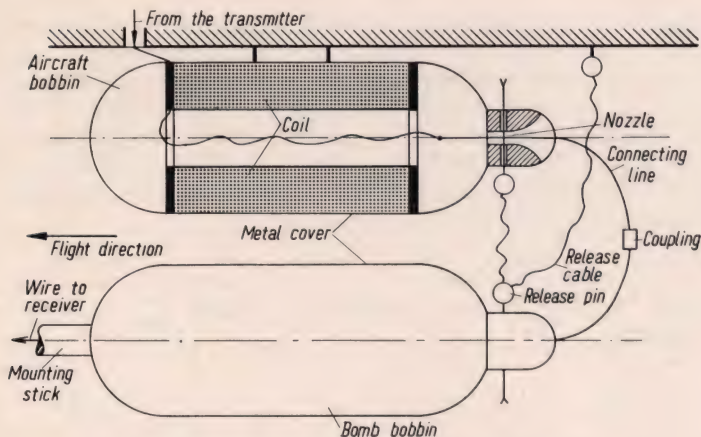


Fig. 4. Installation of the wire-coils for Hs 293

bodies. In order to avoid excessive damping by the coil windings when using audio-frequency transmission, both ends of the layers were bared and short-circuited by metal discs. The bobbins attached to the aircraft carried 12 km of wire, those on the bomb 18 km; hence the total length of wire was  $2 \times 30$  km. The wave resistance of the two-wire line was about  $1200 \Omega$ . FRITZ X carried the bobbins on the two end discs of the tail plane. Because of the direct current transmission, damping did not so much matter, and therefore insulated wire was used without short-circuiting of the windings. In view of the shorter range a total wire length of  $2 \times 8$  km was chosen.

## 5. LINE CONTROL SYSTEM OF THE X-4 AIR-TO-AIR MISSILE

The problem of guiding bombs by remote control was successfully solved by the development of the bombs FRITZ X and Hs 293 in conjunction with the "KEHL/STRASSBURG" radio control system and its later developments. Experience gained in this process gave a complete insight into, and control over, all the various aspects of the problem.

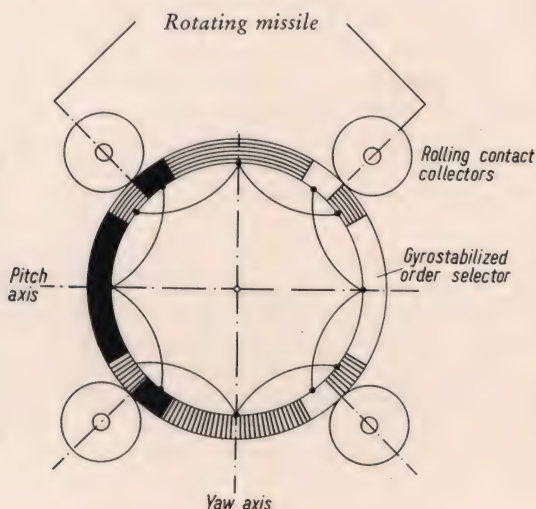
It is not surprising therefore that the development of the war and the loss of air superiority gave rise to the idea of simplifying the missiles, and their control, to such an extent that they could be used as fighter weapons against aircraft. Thus the X-4 and Hs 298 air-to-air missiles came into being.

The X-4, a development of the FRITZ X, was line controlled with the "DÜSSELDORF" (FuG 510) system as transmitter and a system similar to the "DETMOLD" type as receiver. The "DÜSSELDORF" transmitter was essentially the same as the "DÜREN", but its working voltage was only 200 V, as compared with the 600 V of the "DÜREN". There were only two bobbins, and they were fixed at the wing tips of the missile; each of them took 5500 m of insulated steel wire of 0.2 mm diameter. The maximum current was 5 mAmp.

As the X-4 was not gyro-stabilized, but rotated about its roll-axis at approximately one rev./sec, it was necessary to have a gyro-stabilized order selector for the 4 spoiler switching sets. Prior to the release of the missile, the

gyroscope was brought up to speed, via a 7-pin break-off connector, by a 3-phase motor-generator inside the aircraft which was driven from the mains. After the release it continued running under its own momentum. By this arrangement a very simple remote control system was achieved.

Because of the rotation of the missile, the spoilers could only receive desired orders when the corresponding rotating collectors were just going through the pitch and yaw axis of the gyro-stabilized order selector. This gave rise to some steering errors. In order to achieve a better agreement between order and spoiler action, a segmented selector unit, as shown in Fig. 5, was used. Four 60°-segments and eight 15°-segments were distributed over the full circumference. This arrangement ensured an exact coincidence of order and action for every 45° of the rotation on an average. In consequence, it was necessary to raise the switching frequency of the order distributor unit to 20 c/sec.



*Fig. 5. Control distributor for X-4*

No serious difficulties were encountered during the trials, but in order to obtain accurate spoiler change-over without bounce and equal short operating times (10 msec total) the T 57 relay in the missile had to be replaced by the T 64. The bobbins gave some trouble at first, but this was essentially due to flaws in manufacture.

Although some objections were raised against the use of wire control over friendly territory because of the possible interference with road traffic and overhead electricity grid, the X-4 was at first proposed solely for line control. However, when the "KOGGE" control system in the decimetre range was under development, a special receiver ("WALZENBRIGG") was envisaged, which could be fitted into the bobbins of the X-4.

In this connection, although it was not an airborne, but an infantry anti-tank, weapon, the X-7 of Dr. KRAMER should be mentioned. It was a sort of a line-remote controlled "PANZERFAUST" and was the simplest possible device of its kind. Like the X-4, it rotated about its roll axis and had a single cogged ridge

spoiler, which received its orders via a gyro-stabilized order selector. There were separate guide sticks for up-down and left-right control. The trials of this system had not been completed in April 1945, so that it was never commissioned.

## 6. RADIO REMOTE CONTROL SYSTEM OF THE HS 298 AIR-TO-AIR MISSILE

Prof. WAGNER's Hs 298 air-to-air missile, which corresponded to the Hs 293 glider bomb, was radio controlled. The transmitter set was a "KEHL" system Mark FuG 206 and the receiver set used a "COLMAR" receiver and was designated FuG 232. The transmitter was not appreciably different from the standard "KEHL"-system, but the receiver was considerably simplified due to the simple flight control system of the Hs 298. A wind driven generator replaced the battery in the missile. The "COLMAR" receiver was super-regenerative, and its audio frequency stage, similar to that of the "DUISBURG" system, contained only two audio frequency discriminator sets instead of 4 sets of filters, so that it required relatively little space. Corresponding to the shorter range of the missile, the sensitivity of the receiver was only one third of that of the "STRASSBURG" type, viz.  $6 \mu V$ . The channel spacing was 200 kc/sec, so that the entire frequency range was covered with two types of 4 and 5 frequencies respectively.

The simple flight control system of the Hs 298 required only a very simple connecting device between receiver output and flap switching sets. Due to the use of one aileron for control about the roll axis and of a spoiler in place of the elevator, the filter- and rectifier stage of the connecting device were dispensed with, and the signals fed in directly as in the case of FRITZ X. Sectional construction, with the essential parts in separate units, was chosen to facilitate service handling. The "KOGGE" project, in the decimetre range, was also designed in such a way that it could be readily substituted for the "KEHL-COLMAR" system.

## 7. SURVEY OF COMPLETED REMOTE CONTROL SYSTEMS

After having treated in detail the problem of the remote control of airborne bombs and air-to-air missiles by describing the completed and commissioned system, a survey of all the developed systems will now be given. See Table 1. The amount of development work which had to be done under the rigorous restrictions imposed by a war of attrition which, from our point of view, took an increasingly unfavourable course will become strikingly apparent. Four "KEHL-STRASSBURG" systems were developed by the firms TELEFUNKEN, STARU and OPTA for use with FRITZ X and Hs 293. Of these, "KEHL I" was designed for dropping one FRITZ X, "KEHL III" for dropping one Hs 293 and "KEHL IV" was originally intended for the optional dropping of one Hs 293 or one FRITZ X, but later development made optional dropping of one to four FRITZ X's or Hs 293's possible. "KEHL IV" was especially developed for use with the large bombers Do 217, FW 200 and He 177.

The FuG 203/230 h system was an intermediate model with the additional transmission of a fuse-ignition signal on 3.5 kc/sec as a special feature. It was

Table 1. Completed control systems for guided missiles.

J. Dantscher, Further Development of Remote Control System

Transmitters				
System	Code name	Maker	Stage of development	Technical data
FuG 203 a	KEHL I	TELEFUNKEN OPTA	Combat use	48...50 Mc/sec Amplitude modulation. 18 channels. Modulation frequencies 1/1.5/8/12 kc/sec
FuG 203 b	KEHL III	TELEFUNKEN OPTA	Combat use	As KEHL I
FuG 203 c	KEHL IV	TELEFUNKEN OPTA	Combat use	As KEHL I with selector switch for 1 FRITZ X and 1 Hs 293
FuG 203 d	KEHL IV	TELEFUNKEN OPTA	Combat use	As KEHL I with selector switch for 1—4 FRITZ X and 1—4 Hs 293
FuG 203 h	KEHL	TELEFUNKEN OPTA	Prototype	As KEHL I, with 5th modulation frequency (3.5 kc/sec) for fuze ignition
FuG 203-1	KEHL-1	TELEFUNKEN OPTA	Combat use	Alternative frequency 60 Mc/sec. Otherwise as KEHL I...IV
FuG 203-2	KEHL-2	TELEFUNKEN OPTA	Combat use	Alternative frequency 27 Mc/sec. Otherwise as KEHL I...IV
FuG 205	GREIFSWALD	LORENZ	Prototype	48...50 Mc/sec. Frequency modulation. 18 channels
FuG 206	KEHL	TELEFUNKEN	Prototype	4 modulation frequencies as KEHL (?) As KEHL I, with 5th modulation frequency (3.5 kc/sec) for fuze ignition
FuG 207	DORTMUND	STARU (TELEFUNKEN)	Combat use	Audio frequency two-wire line transmission. Phase-shift free modulation with frequency-switching 422/473 and 665/735 c/sec. Coils at both ends
FuG 208	DÜREN	STARU (TELEFUNKEN)	Manufacture	D.C. two-wire line transmission. 600 V. Switch-over with relay guide stick (plus-minus, high-low). Coils at one end only
FuG 510	DÜSSELDORF	DONAG (TELEFUNKEN)	Manufacture	D.C. two-wire line transmission. 200 V. Coils at one end only
FuG 512	KOGGE	TELEFUNKEN	Prototype	1100 Mc/sec. Replacement for KEHL. Phase-shift free modulation with frequency-switching 6/9 and 13/16 kc/sec. Preliminary transmitter model KAI, Serial model KRAN. Flip-flop guide stick "KLAPPER" with "KARTE" and "POL" device. Fuzing signal by 200 c/sec switching frequency
Receivers				
System	Code name	Maker	Stage of development	Missile to which applied
FuG 230 a	STRASSBURG	STARU	Combat use	1 FRITZ X
FuG 230 b	STRASSBURG	STARU	Combat use	1 Hs 293
FuG 230 a and b	STRASSBURG	STARU	Combat use	1 FRITZ X and 1 Hs 293
FuG 230 a and b	STRASSBURG	STARU	Combat use	1—4 FRITZ X and 1—4 Hs 293
FuG 230 h	STRASSBURG	STARU	Prototype	Hs 293 and 298
FuG 230-1	STRASSBURG-1	STARU	Combat use	FRITZ X and Hs 293
FuG 230-2	STRASSBURG-2	STARU	Combat use	FRITZ X and Hs 293
FuG 235	KOLBERG	LORENZ-STARU	Prototype	FRITZ X and Hs 293
FuG 232	COLMAR	FRIESECKE & HÖPFNER	Prototype	Hs 298 (and X-4)
FuG 237	DUISBURG	STARU	Combat use	Hs 293 (and FRITZ X)
FuG 238	DETMOLD	STARU	Manufacture	FRITZ X (and Hs 293)
FuG	(DETMOLD)	DONAG	Manufacture	X-4 (and X-7)
FuG 530	KOGGE (BRIGG and FREGATTE)	TELEFUNKEN	Prototype	FRITZ X, Hs 293, Hs 298 and X-4

Table 2. Control receivers for guided missiles.

J. Dantscher, Further Development of Remote Control Systems

System	Code name	Carrier	Principle	Channels	Modulation	Other technical data	Maker	Stage of development	Missile to which applied
E 30	STRASSBURG	48 ... 50 Mc/sec	Super-heterodyne	—	Amplitude modulation 1/1.5/8/12 kc/sec	Output: switch valves RL 12 P 10 S	STARU	Preliminary series	FRITZ X and Hs 293
E 230	STRASSBURG	48 ... 50 Mc/sec	Super-heterodyne	18 channels, channel spacing 100 kc/sec	Amplitude modulation 1/1.5/8/12 kc/sec	Output: relay T.rls. 64; valves RV 12 P 2000, RV 12 P 2001, RG 12 D 2. sensitivity $\approx 2 \mu V$ , automatic volume control 1:10 <sup>5</sup>	STARU	Combat use	FRITZ X and Hs 293
E 230 h	STRASSBURG h	48 ... 50 Mc/sec	Super-heterodyne	As E 230	Amplitude modulation 1/1.5/8/12 kc/sec 3.5 kc/sec as fuze ignition signal	As E 230	STARU	Prototype	Hs 293 and 298
E 230-1	STRASSBURG-1	$\approx 60$ Mc/sec	Super-heterodyne	9 channels, channel spacing 200 kc/sec	As E 230	As E 230	STARU	Combat use	FRITZ X and Hs 293
E 230-2	STRASSBURG-2	$\approx 27$ Mc/sec	Super-heterodyne	9 channels, channel spacing 200 kc/sec	As E 230	As E 230	STARU	Combat use	FRITZ X and Hs 293
E 231	MARBURG	48 ... 50 Mc/sec	Super-heterodyne	As E 230	Amplitude modulation 2 audio frequency discriminators for phase-shift free modulation. 5 modulation frequencies Frequency modulation, otherwise as E 230	—	STARU	Prototype	Hs 293
E 235	KOLBERG	48 ... 50 Mc/sec	?	18 channels	Frequency modulation, otherwise as E 230		LORENZ	Prototype	FRITZ X and Hs 293
E 232	COLMAR	48 ... 50 Mc/sec	Superregenerative	4 and 5 channels channel spacing 200 kc/sec	5 modulation frequencies, otherwise as E 230	Sensitivity $\approx 6 \mu V$	FRIESECKE & HÖPFNER	Manufacture	Hs 298 (and X-4)
E 237	DUISBURG	two-wire-line	Audio freq.	—	2 audio frequency discriminators 422/473 and 665/735 c/sec	Automatic volume control 1:10 <sup>5</sup> Switch valves RG 12 D 2	STARU	Combat use	Hs 293 (and FRITZ X)
E 238	DETMOLD	two-wire-line	D.C.	—	plus-minus/high-low	2 relays T.rls. 64, 2 rectifiers	STARU	Manufacture	FRITZ X (and Hs 293)
E 530	BRIGG	$\approx 1100$ Mc/sec	Superregenerative	18 channels (?)	2 audio frequency discriminators for phase-shift free modulation, 6/9 and 13/16 kc/sec. Fuse ignition signal from 200 c/sec switching frequency		TELEFUNKEN	Prototype	FRITZ X, Hs 293
	WALZENBRIGG	$\approx 1100$ Mc/sec	Superregenerative	?			TELEFUNKEN	?	Hs 298, X-4
	FREGATTE	$\approx 1100$ Mc/sec	Super-heterodyne	18 channels	As BRIGG		TELEFUNKEN	Prototype	FRITZ X, Hs 293

intended for use with the Hs 298 for testing purposes and for dropping the Hs 293 into bomber formations. The systems FuG 203/230 -1 and -2 were developed in case the types mentioned above were put out of action by effective enemy jamming. They were held in store for this emergency, but it did not arise and therefore they did not go into active service.

The "GREIFSWALD-COLBERG" system was a parallel development by the LORENZ Company using frequency modulation. It too was planned to serve as a substitute in the case of jamming, but the project was abandoned in the later stages of development.

The system FuG 206-232 with the "COLMAR" receiver had an additional (fifth) modulation frequency and was developed for radio control of the air-to-air missiles, but it did not come into service, as the Hs 298 was abandoned in January 1945.

The next two systems are the line control systems. The "DORTMUND-DUISBURG" audio frequency system was used finally for the glider bomb Hs 293. The "DÜREN-DETMOLD" direct current system was developed to guide the FRITZ X, and a variant of it, the "DÜSSELDORF-DETMOLD" direct current line control system, was used with the air-to-air missile X-4.

This line of development came to a conclusion with the "KOGGE" or FuG 512/530 radio control system which made use of all the experience obtained with its predecessors. It worked on a wavelength of about 27 cm, but it did not reach a sufficiently advanced stage to go into production. Difficulty was experienced in maintaining the frequency, energy, and synchronism, of the transmitter prototype "KAI", over the entire band. For this reason a new model "KRAN" was developed and, to my knowledge, went into experimental production.

## 8. SURVEY OF REMOTE CONTROL RECEIVERS

Table 2 gives a survey of the receivers developed for the remote control of airborne bombs. The prototype model E 30 was not equipped with a locking channel selector, was less sensitive, and in many other respects was much simpler than the E 230. This model was used mainly in the first stages of the tests. The production model "STRASSBURG" E 230 had 18 channels, with locking selectors, in the frequency range 48 to 50 Mc/sec and a channel spacing of 100 kc/sec. It used amplitude modulation with 1, 1.5, 8 and 12 kc/sec; the sensitivity was  $2\mu\text{V}$  and the automatic volume regulation  $1/10^5$ . In the output were two relays T.rls. 64. All the drops of FRITZ X and the Hs 293 were carried out with this receiver. The receiver E 230 h responded to a fifth modulation frequency of 3.5 kc/sec which gave a fuse ignition signal, as mentioned above. The receivers E 230-1, -2, working on longer and shorter wavelengths, were held in reserve as substitutes in the case of enemy jamming.

On switching the various modulation frequencies, the random phase angle shift between the two frequencies causes voltage leaps, if the modulation frequencies are produced by separate generators.

These voltage leaps cause a certain jitter in the transmission of orders and the accuracy is reduced. In order to eliminate the phase angle shift error, a modulation switch-over was developed which used a single modulation generator

and varied its frequency by switching an additional inductance on and off, thus increasing and decreasing its effective  $L \times C$  value. This type of frequency generation is shown in the schematic frequency-switching diagram of the "DORTMUND-DUISBURG" system (Fig. 2). The receiver "MARBURG" was developed to receive the switched frequencies. It accepted them with 2 discriminators and converted them through relays. The output stage of this receiver corresponded to the receiver of the "DORTMUND-DUISBURG" system.

For the F. M. high frequency system a receiver, "KOLBERG", was developed by the LORENZ Company. Apart from the difference in modulation it was built on the lines of the "STRASSBURG", but its production was not taken up owing to the lack of industrial capacity.

The "COLMAR" receiver was developed for the air-to-air missiles. It was a super-regenerative receiver covering, with two models of 4 and 5 channels respectively, the total frequency range. The channel spacing was 200 kc/sec. Its sensitivity was only one third of that of the "STRASSBURG". Built by FRIESECKE & HÖPFNER, it was used with the Hs 298.

The receivers "DUISBURG" and "DETMOLD" were developed for the two-wire line control. "DUISBURG" as an audio frequency receiver consisted of two discriminators for switching without phase angle shift and had an automatic volume regulation of  $1/10^5$ . It was assigned to the Hs 293. "DETMOLD" was the direct current receiver and consisted merely of two rectifiers and two T. rls. 64 relays. By switching over the four states: plus, minus, high, low, the two qualified rudder orders were transmitted.

Before the end of the war two receivers were developed for use with the "KOGGE" decimetre transmission system. The first model, "BRIGG", was super-regenerative and worked on a wavelength of about 27 cm. In its audio frequency stage it had two discriminators for the four modulation frequencies 6/9 and 13/16 kc/sec. A fifth order, an ignition order for operating the fuse, was given by means of the switching frequency of 200 c/sec, whereas the standard switching frequencies were the same as used in FRITZ X and Hs 293, viz. 5 and 10 c/sec. A special variant of this receiver, the so called "WALZENBRIGG", was intended for use in the air-to-air missiles. Its dimensions corresponded to those of the bobbins of the X-4. This made it possible to exchange it for the wire coil in the case of a change-over to radio control. As a final solution, the "FREGATTE" superheterodyne receiver was developed.

The two tables shown convey a good picture of the extent of development work which had to be undertaken by industrial firms for the benefit of the few types of remotely controlled bombs produced during the last war. The results are all the more worthy of notice, as they had to be achieved in the face of other heavy commitments imposed by a modern technical war, which, on the other hand, crippled industry directly and indirectly to an ever rising degree by enemy action. It was the pressure of wartime requirements which forced the development to spread in breadth right from the outset to an extent which in peace time would have been necessary only after its final conclusion as preparation for the equipment of the service.

# A GUIDING SYSTEM USING TELEVISION

FRITZ MÜNSTER\*

## 1. INTRODUCTION

Using as a starting-point the Hs 293 tele-guided bomb, developed by Prof. H. WAGNER at the HENSCHEL Aircraft factory, a special development programme was carried out at the same plant in which television was used for the first time as a means of guiding a missile. The television gliding-bomb was intended for the same purpose as the Hs 293 and used identical equipment for the flight-control; nevertheless the application of television for picking up the target raised a number of new problems, particularly in connection with the control technique.

Since in the case of the missile a television camera was to pick up the image of the target and the bomb-aimer in the carrier-aircraft was to guide the gliding bomb according to the target-image transmitted to him, the basis for locating the target was transferred from the control stand in the aircraft to the missile itself. Thus, the television-guiding method retained typical features of the homing system. Whereas, however, the control function chosen for a target locating procedure can be expressed by mathematical formulae, because the bomb-aimer is involved the television-guiding procedure must be considered as a psychological process.

Since they were fundamentally new at that time and are interesting from a retrospective point of view, attention should be drawn to two points:

- a) the development and application of the television equipment,
- b) the control technique which, because of the different subjective impression upon the bomb-aimer and the reactions required of him, needed modification.

In this paper the stage of development will be dealt with in which the Hs 293 D was actually tested. Later on, intended improvements which, however, did not pass the planning stage, will be discussed.

## 2. TACTICAL CONSIDERATIONS

### 2.1. Attack Procedure Using the Coincidence Method

Some aspects of the stage already reached should be explained first, since they were essential for all further improvements. The Hs 293 gliding-bomb had been developed mainly for use against sea-targets. It was dropped from a carrier-

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aircraft the bomb-aimer of which guided it towards the target. For the control signals, the "KEHL-STASSBURG" equipment was used.

The bomb-aimer operated his control stick so as to keep the bomb on a trajectory defined by continuous optical coincidence with the target (the so-called three-point-coincidence method), during which time the carrier-aircraft had to follow a prescribed course.

Coincidence of bomb and target was maintained visually by the bomb-aimer and the bomb carried a tracing light so that it was easily discernable. The carrier-aircraft had to continue on a fixed flight path after the release of the bomb for two reasons (see Fig. 1):

- a) the bomb-aimer, seated on the starboard side of the cockpit, had to have an unobstructed view of the target during the guiding manoeuvre,
- b) observation and keeping track of bomb and target would become rather difficult if the carrier-aircraft were allowed to change its course.

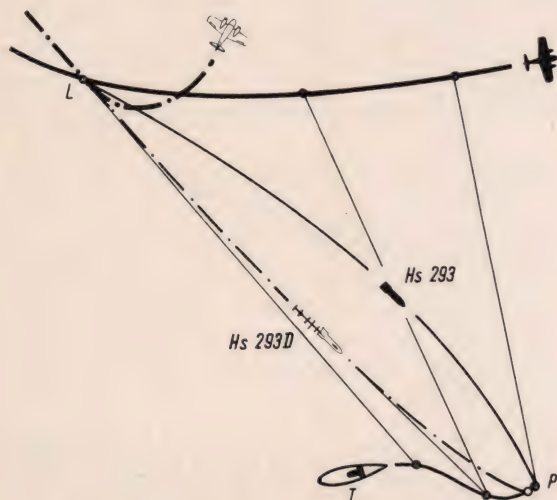


Fig. 1. Carrier-aircraft flight path restricted for launching Hs 293

L = Launching point, P = Impact point, T = Target,  
—— Line-of-sight course, - - - Pursuit course.

Naturally, at large distances the target would appear rather small while, in the terminal phase of the attack, control of the bomb, to such a sensitive degree as to hit a particular part of the ship, would become more or less impossible. In addition, conditions of visibility are in many cases unfavourable for not only geometrical, but also meteorological reasons, owing to layers of haze in the atmosphere. Furthermore, some peculiarities of the acceleration-control system presented difficulties which will be explained later when discussing the control technique.

## 2.2. Tactical Advantages of Guiding by Television

With regard to the tactical restrictions resulting from the necessity for the carrier-aircraft to follow a fixed course and also from the reduction of bombing

accuracy at large distances, the application of television at first promised the following advantages:

- a) Since the bomb-aimer guides the bomb according to the image presented to him on the screen and no longer needs to see the target himself, and since also the trajectory of the bomb no longer depends on the course of the aircraft, all restrictions on the course of the aircraft once the bomb has been released disappear. The aircraft can evade anti-aircraft fire and may even hide in the clouds. The bomb-aimer, so to speak, will guide his bomb to the target as he returns home.
- b) The target-image on the T. V. screen grows larger as the bomb approaches the target and becomes, due to atmospheric conditions, more distinct, so that an accurate hit on a vulnerable part of the target may be more easily achieved.

Moreover, additional advantages in control technique appear.

### 3. TECHNICAL CONSIDERATIONS

#### 3.1. State of Development of Television

It was possible to base the development of the special television equipment required on the achievements which, prior to World War II, had led to the experimental introduction of a T. V. broadcasting system. T. V. standards at that time were based on a picture with 441 lines and the method of interlaced scanning. The picture speed was 25 per second for the total image consisting of two parts. The quality of reproduction was rather good and appeared fully adequate for the planned guiding system.

#### 3.2. Special Requirements

Since commercial T. V. equipment is normally built for stationary use and is subject to few restrictions as to the admissible amount of equipment and transmitting energy, a new development programme had to be launched to cope with the entirely different conditions for the use of such apparatus in a missile.

In the main, the new apparatus had to fulfil the following conditions <sup>1</sup>:

1. small size and light weight of the equipment,
2. low power consumption,
3. simple design and use of only a few types of valves,
4. no operational adjustments on camera and transmitter,
5. few adjustment-requirements on the receiver side,
6. insensibility to shock and vibration,
7. insensibility to altitude,
8. insensibility to climatic conditions.

### 4. PRACTICAL SOLUTION

#### 4.1. Description of the T. V. Equipment <sup>2,3</sup>

The T. V. equipment, consisting of the "TONNE A" camera, the transmitter, and the "SEEDORF" receiver, was developed by the FERNSEH-GMBH. in co-

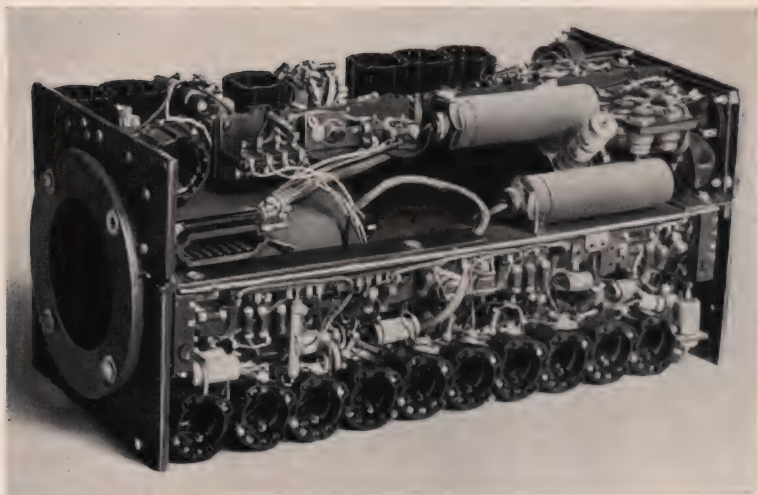
operation with the research laboratory of the German Post Office. The method of interlaced scanning was dropped and, in view of the fast movements involved, a picture frequency of 50 per second with 224 lines was chosen. Since the target usually presented its main dimension horizontally, the better optical resolution was required in this direction; therefore, after some tests, the lines were posed vertically.

#### *4.1.1. Camera*

The SUPER ICONOSCOPE IS 9 was chosen as picture storage tube in the camera (see Fig. 2). The approximately  $7 \times 9$  mm optical image, which was focussed on a Caesium photocathode, was electronically projected on to a mosaic plate by a magnetic lens on a 5:1 scale, where it was scanned by an electronic beam.



*Fig. 2. Iconoscope IS 9*



*Fig. 3. TV-camera "Tonne"*

The optical lens was the ZEISS "BIOGON" which had a focal length of 35 mm and a relative opening of  $F = 2.8$ . The resulting image angle was about  $\pm 13^\circ$ .

The camera required the following accessories:

a pulse generator,

two sweep generators (for lines and for total image resp.),

a VIDEO amplifier (Fig. 3).

These components were assembled as self-contained sets. They were built, together with the storage tube, into a common chassis of size  $17 \times 17 \times 40$  cm. This equipment fulfilled the requirements of compactness and small size very well. Power was supplied by a 24 volts DEAC Battery through an OEMIG Converter of 38 watts supplying a feeder set with A. C. at 500 c/sec.

The camera required 29 valves of only two different types: 27 RV 12 P 2000 and two RL 12 T.

By using only two types of valve and breaking down the camera into several components which were plugged together, the third precondition was well complied with.

To fulfil the fourth precondition, i.e. the omission of operational adjustments on camera and transmitter, special measures were taken regarding the power-supply of the image storage tube:

- a) The heating current of the tube was stabilized by an Iron-Hydrogen resistor.
- b) The anode voltage of 800 V for the scanning beam was kept constant by neon stabilizer tubes.
- c) Electronic D. C. stabilization was arranged for focussing the scanning beam and for feeding the magnetic lens.

Due to these stabilizing measures in conjunction with temperature compensation in some circuits, re-adjustments became wholly unnecessary.

Furthermore, a D.C. feedback was applied with subsequent automatic gain control of the amplifier. Thus, the whole range of modulation available for the transmission of the optical image could be used regardless of the brightness of the targets. Any additional adjustment of a diaphragm or of the amplifier gain could therefore be spared. An automatic diaphragm control which had been envisaged, proved however to be superfluous. Thus, the camera was insensible to changes in the average light intensity. This resulted, together with the voltage stabilization, in the complete elimination of any need for camera adjustments.

Tests of the insensibility of the camera to shock and vibrations gave satisfactory results.

The camera as tested was not exactly altitude-proof, yet it stood up to all demands encountered during flying tests.

The stabilization of the main voltages and the temperature compensation of the components resulted in the camera being indifferent to temperature changes within wide limits. A practical test carried out whilst the air-moisture was high, showed that the camera was also insensitive to climatic conditions.

#### *4.1.2. Transmitting Technique*

Several new ideas were incorporated in the development of the transmission system. Negative modulation was applied to the carrier frequency, which method has become the standard of the T. V. broadcasting technique of to-day

owing to its advantages. This system supplies a better image with low reception field intensity, since disturbances cause only dark spots instead of bright blurs which result in reducing the clearness of the image. A simple fading regulation can be used, as the synchronizing pulses become positive.

Of particular importance was the synchronizing system chosen. As the safest method, a lock-in synchronization was applied. The camera was fitted with a frequency-stabilized master-generator delivering the line-sweep frequency. The positive synchronizing pulses superimposed on the image pulses govern an identical frequency generator in the receiver. The line sweep on the receiver side is thus supplied by pulses of the frequency generator in the receiver, so that the loss of some pulses from the camera will not discontinue the line sweep in the receiver. By this means the sweep synchronization of transmitter and of receiver was assured even in the event of transmission disturbances.

Transmission of pulses for synchronizing the picture frequency of 50 c/sec was rendered difficult by the lay-out of circuits for such a low frequency. Therefore, this kind of transmission was abandoned and instead the picture frequency pulses were formed in the camera as well as in the receiver by dividing the line-sweep frequency.

Satisfactory stability of the frequency generators was obtained by the use of temperature-compensated circuits achieving an accuracy of  $10^{-3}$  in frequency over a temperature range from  $-10^{\circ}$  to  $+70^{\circ}$  C.

The method of lock-in synchronization used had above all the great advantage that no disturbances were caused by the loss of synchronizing pulses. Thus, any serious deterioration of the image by the loss of some lines or groups of lines could be avoided.

The television transmitter had an output of approximately 20 watts with a frequency of about 400 Mc/sec, i.e. a wavelength of 73 cm, and a 60% modulation range for transmitting. A YAGI aerial with a reflector, a radiating element, and three wave guides were used. Power was supplied by an OEMIG converter, fed by the same battery that supplied the camera, and delivering A.C. of 60 W at 500 c/sec.

#### *4.1.3. Receiver*

For receiving the image the "SEEDORF" receiver had been developed (Fig. 4). It used a cathode-ray tube with a screen diameter of 13 cm. The accelerating voltage was 6 kV. The image tube was built, together with the output stage of the amplifier, the sweep generators and the high-tension supply circuit, into one chassis. The high-frequency part and the pulse generator with the frequency divider were mounted on a separate chassis which could be plugged to the other unit.

The size of the complete set was  $17 \times 22 \times 40$  cm. The screen itself measured  $8 \times 9$  cm.

Constant sharpness of the electronic beam was achieved by stabilizing the high tension voltage and the focussing current.

The range between black and white was automatically controlled and changes in the reception field intensity were also automatically compensated.

These measures made it possible to set up the receiver with only three adjustments, which could even be made prior to the release of the bomb, so that

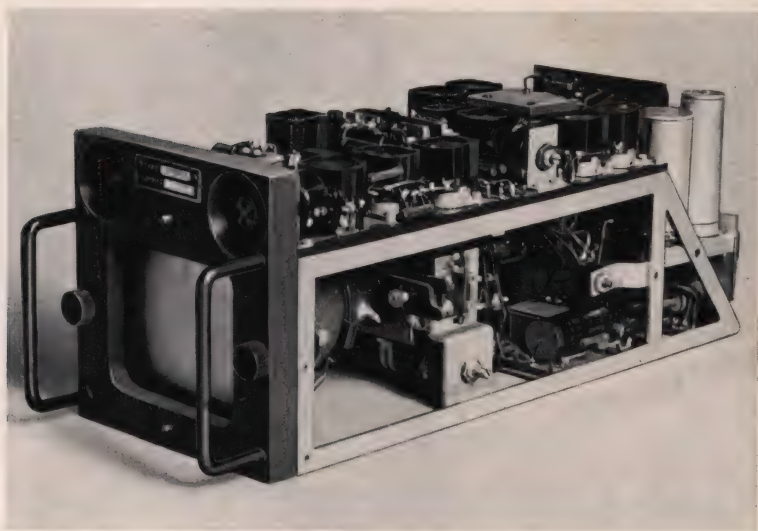


Fig. 4. TV-receiver "Seedorf"

during the guiding manoeuvre no further adjustments were necessary. These were:

- a) Adjustment of the screen brightness by turning the knob on a potentiometer which changed the WEHNELT bias.
- b) Adjustment of the contrast range by turning a potentiometer knob affecting the amplifier characteristics.
- c) Adjustment of the picture phase by push-button operation, interrupting the governor action on the frequency generator.

## 4.2. Installation of Equipment

### 4.2.1. Structural Alteration of the Missile

Rebuilding the Hs 293 for use with television equipment proved not particularly difficult: to the equipment already installed in the bomb the television set had to be added.

The camera, enclosed in a cast frame of magnesium alloy, replaced the trim weight formerly used at the front end of the bomb (Fig. 5). The power set was attached to the underside of the frame by electric plug-racks. The converter was placed between the upper mountings of the cast frame. A wind vane was suspended in a cap centred to the cast frame and forming an aerodynamic fairing over the camera set. The wind vane was connected to the camera by gear segments adjusting the position of the optical lens in the direction of the vertical axis of the missile.

Wind tunnel tests with this arrangement had shown a linear relation between the angle of attack and the displacement of the wind vane, so that a fixed gear-ratio could be applied. Moreover, the wind tunnel tests had shown that the wedge-shaped vane was sufficiently damped.

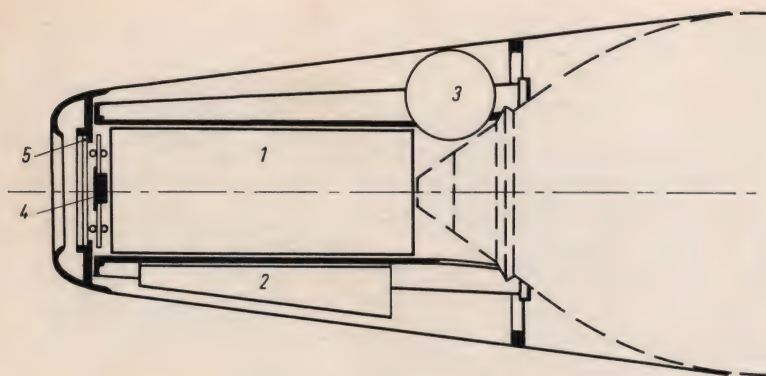


Fig. 5. Camera installation

1 = TV-camera, 2 = Power operating set, 3 = Converter, 4 = Movable objective, 5 = Heated anti-dim glass.

At the front end a clear-sight screen, heated by the battery to avoid condensation and icing, was built in.

The length of the front section of the bomb increased by about 450 mm due to the installation of the camera set. The general arrangement is shown in Fig. 6.

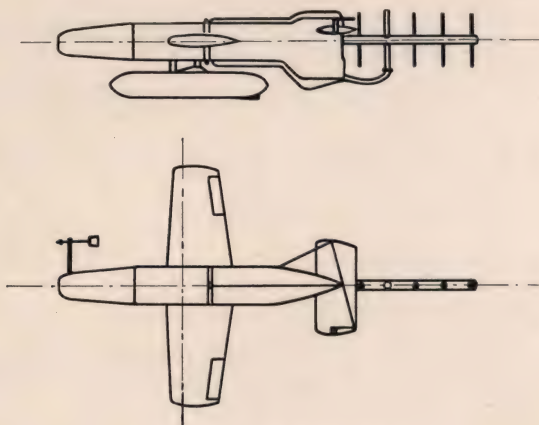


Fig. 6. ASM Hs 293 D

The bombs intended for practical tests were fitted with a longer instrument rack to accommodate in addition to the equipment already installed a DEAC battery, an OEMIG converter and the T.V. transmitter. For this purpose, the rear section of the bomb had to be lengthened by about 230 mm.

The YAGI aerial was placed at the spot where previously the tracing light had been mounted. Below the aerial a trim weight was fitted. In action the tracing light would no longer be required, but it was provisionally fitted to test bombs to facilitate the tracing of the trajectory during test droppings.

The whole conversion increased the flying weight of the bomb by something between 130 and 150 kg.

#### 4.2.2. Conversion of the Carrier Aircraft

At the bomb-aimer's position in the aircraft the "SEEDORF" receiver had to be installed above the bomb control set which was positioned horizontally. The power supply was located in the centre of the fuselage.

The YAGI aerial used for reception had, following the first tests, to be removed to another place to avoid disturbances by propeller modulation.

For the tests, a corrective airstream deflecting surface was mounted beneath the wing of the aircraft, so that the wind vane of the bomb had already picked up the true flight direction before the bomb was released. Thus, there was no need to cage in the wind vane prior to the release of the bomb.

Two converted He 111 and later one converted Do 217 were available for the tests.

### 5. GUIDING TECHNIQUE

With the technical equipment described above the guidance of a missile has now become possible. Before discussing the guiding of the bomb, however, attention has to be drawn to a classification of the guiding problems.

The bomb-aimer guides the missile according to the television image in such a way that the optical axis of the camera always maintains a certain direction in relation to the target. He is given the simple task of directing the optical axis at the target. The guiding technique required will be discussed later. At first it will be assumed that he fulfils his task satisfactorily.

The second problem is that represented by the flight paths along which the bomb-aimer guides his missile. Pursuit curves, which vary according to the direction of the optical axis of the camera in relating to the missile, result.

#### 5.1. Discussion of Flight Paths

The DEUTSCHE FORSCHUNGSANSTALT FÜR SEGELFLUG at Ainring carried out in 1943 a series of theoretical investigations and model tests to determine the behaviour of missiles flying pursuit tracks. These investigations were to examine different systems connected with the installation of the television camera in the missile and the consequent direction of its optical axis. Each of these systems leads to a pursuit curve showing peculiar characteristics<sup>5-8</sup>.

##### 5.1.1. Flight-Path-Fixed System

If the camera axis is continuously kept in direction of the flight-path of the missile, the simplest form of pursuit curve will emerge, where the flight-path tangent is invariably directed towards the target. Theoretical investigations of this curve show the following peculiarities (see Dr. FISCHER's paper in this book, page 28, Fig. 3):

- a) The target will always be hit from the rear, i. e. the path of the pursuer must merge tangentially from the rear into the path of the target.
- b) If the speed of the pursuer is more than twice the speed of the target, the radius of curvature of the missile path will decrease during the approach and become zero at the point of impact.

Since the curvature of its flight-path is limited, the missile will, at the point where the radius of curvature attains its feasible minimum, leave the pursuit track and continue on a circular path of the smallest possible radius. A hitting error will result from this behaviour which depends partly on the flying properties of the missile and partly on the position at the beginning of the attack, i.e. from the initial data of the pursuit curve. As would be expected, the most favourable positions for attack were from the rear, where flight and target velocity vectors formed an angle of less than  $90^\circ$ .

Under battle conditions likely to be encountered at that time, hitting errors up to 16 m had to be faced under unfavourable circumstances. With well-chosen attack conditions hitting errors would become smaller.

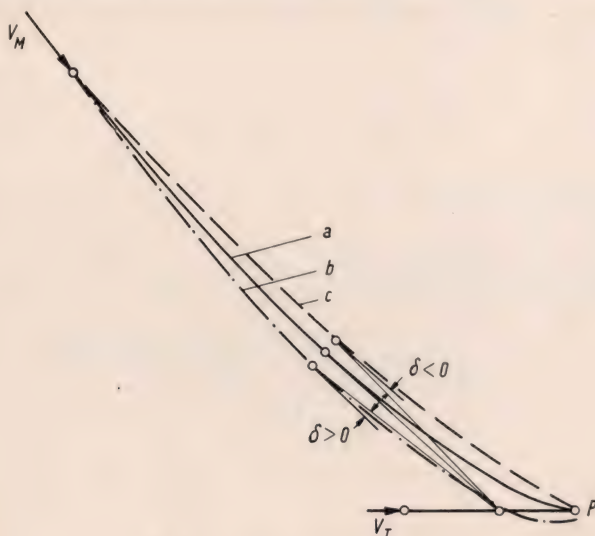


Fig. 7. Deviated pursuit courses

$a$  = Pure pursuit course,  $b, c$  = Deviated pursuit courses.  
 $V_M$  = Missile speed,  $V_T$  = Target speed,  $P$  = Impact point.

These considerations applied to proper pursuit curves. They require the optical axis of the television camera always to be held in the direction of the flight path of the missile. If the optical axis lies at an angle to the flight path direction, the characteristics of the pursuit curve will change (Fig. 7). For a positive deviation angle  $\delta$ , i.e. when the camera axis is deflected towards the inner side of the path curve, the terminal curvature will be tighter than in the original pursuit curve  $a$ . A negative deviation  $\delta$  has the opposite effect, i.e. a tighter curvature at the beginning, easing up towards the end of the flight path  $c$ . Both possibilities will be dealt with more fully.

### 5.1.2. Missile-Fixed System

A missile-fixed system is obtained when the television camera is rigidly mounted with its optical axis parallel to the longitudinal axis of the missile. In this case the tangent to the flight path will deviate from the direction to the target by the angle of attack and by the angle of yaw. This corresponds to the

curve  $b$  of Fig. 7. The pursuer will reach the point of tightest possible curvature earlier. The pursuit curve being left earlier, the hitting error described before will increase considerably. But even if the possible curvature were unlimited, the pursuer would not hit the target, but would spiral around it without ever reaching it.

The missile-fixed system, though uncomplicated regarding the installation of the television camera, is not suited for guidance along a pursuit track.

### 5.1.3. Constant Bearing System

Before considering a system with negative deviation which seems to promise a more advantageous flight path, it may be useful to pay some attention to the constant-bearing system (Fig. 8).

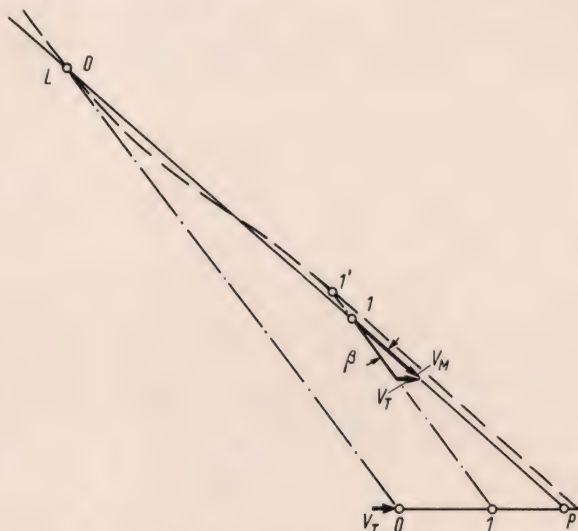


Fig. 8. Constant bearing course

$L$  = Launching point,  $V_M$  = Missile speed,  $V_T$  = Target speed,  $P$  = Impact point,  
 $\beta$  = Lead angle.

— · — Gyro axis = Line of sight.

From each release point a straight flight path to the target can be defined, if the target is moving straight onwards. It is evident from the velocity vector diagram, that the line of sight is moving parallel to itself in space. If, previous to the release, the optical axis is directed on the target and then held in this direction by a stabilising gyro, it will only be necessary to guide the missile in such a manner that the optical axis points continually towards the target, i.e. to establish permanent coincidence of gyro axis and line of sight. Thus, the parallel shift of the line of sight will be obtained and the straight collision course realized.

The carrier aircraft may approach in the direction of the target, whilst the gyro axis, which is coupled to the optical axis, is still fixed parallel to the longitudinal axis of the aircraft. With the release, when the guiding manoeuvre starts, the gyro will be released.

Fig. 8 shows that with this approach a change of direction equal to the lead angle  $\beta$  must be made with zero radius of curvature at the beginning of the guiding manoeuvre. However, it is only possible to turn in a circle of finite radius and so not point 1 on the original collision course, but the point 1' on a new collision course shifted parallel to the original one is reached. Two flight-paths of opposite curvature, requiring two control signals opposing each other, will thus have to be generated. Such dual command signals will also have to be given in case of further deviations from the collision course in order to guide the missile back to its original course. It will be shown later on, when dealing with the guiding technique, that such a control method is not advisable owing to the extreme skill required from the bomb-aimer.

#### 5.1.4. Inverse Rotation System

A compromise which reduce the difficulties of both constant-bearing-tracking and pursuit-tracking, is offered by a combination of these two methods (Fig. 9).

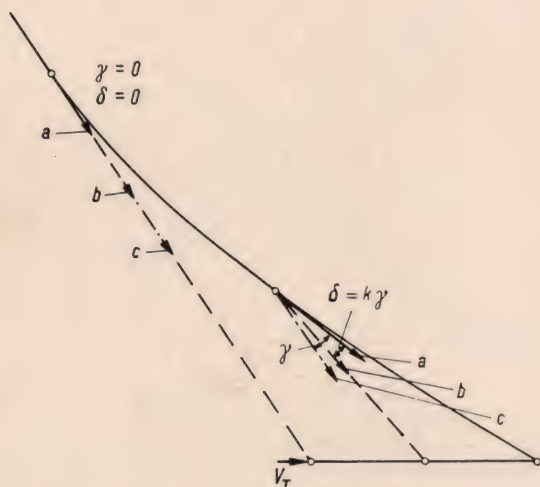


Fig. 9. Course with inverse rotation

$a$  = Flight direction,  $b$  = Line of sight,  $c$  = Gyro axis,  $V_T$  = Target speed.

To achieve this, the optical axis is given a direction lying between the flight path tangent and the direction of the gyro axis, which represents a line fixed in space. The angle  $\delta = k\gamma$  may be defined as the deviation of the optical axis from the flight path direction;  $\gamma$  is the angle between the gyro axis and the tangent to the flight-path, and the constant  $k$  may be altered to give different flight-paths. It will be noticed that

$k = 0$  gives the pursuit curve,

$k = 1$  gives the straight collision course.

By choosing the correct value for  $k$  a flight-path is obtained which begins as a dog curve with a tight though controllable curvature which eases up into an almost straight flight with a constant lead angle. This corresponds to the

previously mentioned effect of a negative deviation angle  $\delta$  between optical axis and flight direction for which a less severe curvature was to be expected.

The straight-line characteristics prior to the hit still hold valid, when a combination of the gyro-fixed and missile-fixed systems in place of the flight-path-fixed system is adopted, since the rather small angle of attack does not cause errors of any consequence during the almost straight course.

Since any rotation of the missile causes an inversely directed rotation of the optical axis in relation to the missile axis, this system used to be called the inverse-rotation system.

#### 5.1.5. *Choice of System for the Hs 293 D Missile*

Consideration of the systems discussed and of the relevant flight-paths shows that neither the missile-fixed system nor the constant-bearing system was suited for the television gliding bomb. While the inverse-rotation system was postponed for later development, it was decided to use the flight-path-fixed system for the Hs 293 D.

The angle of attack was regulated by aid of the wind vane. Since no rudder control was provided which might have caused large angles of yaw, a yaw compensation was omitted. Any angles of yaw still occurring were considered negligible and in fact did not disturb the guiding control.

### 5.2. Discussion of the Control Methods

#### 5.2.1. *Disadvantages of Three-Point-Coincidence Tracking*

Since now, after having discussed the various possible pursuit curves, the guiding procedure is to be discussed, it might be useful to refer to the guiding control of the Hs 293 since the same control equipment was used without any alteration in the Hs 293 D.

The bomb-aimer operated a stick on his control set which sent out corresponding signals by the "KEHL" signal transmitter; these signals were picked up by the "STRASSBURG" control receiver and transferred to the steering mechanism. A rotation of the stick around its zero axis made the bomb roll for an angle of the same sign and size as imposed on the control stick by affecting the lateral roll stabilizer of the bomb. A radial deflection of the stick affected the elevator control of the bomb so as to create a load factor proportional to the stick deflection.

These two control signals met the requirements of the Hs 293 missile, since it was controlled by elevator and aileron only around the transverse and longitudinal axes respectively. The system represented a typical acceleration control with an acceleration of the missile perpendicular to its path and proportional to the radial deflection of the control stick.

It is essential here to explain in detail some peculiarities of the acceleration control method (Fig. 10). When the bomb-aimer notices that the bomb and target do not coincide he has to impose, in accordance with what he observes, a transverse acceleration on the bomb in the direction of the target.

Starting from the stick position prevailing at this moment, he has to apply an additional stick deflection according to the desired corrective transverse movement of the bomb. The new position of the stick has to be maintained until

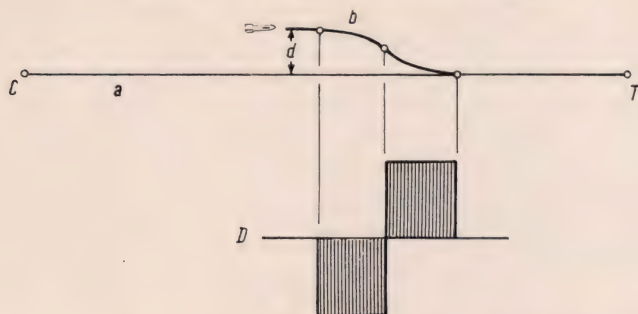


Fig. 10. Doubled command for line-of-sight courses  
 $a$  = Line of sight,  $b$  = Missile flight path,  $d$  = Deviation,  
 $C$  = Command post,  $D$  = Joystick deflection,  $T$  = Target.

shortly before the bomb again coincides with the target, and then a counter-signal, by opposite deflection of the stick, has to be given to avoid overshooting the target. Thus, it is evident that for every correction two opposite signals must be given.

(It may be remembered here, that guiding with a constant bearing system on a lead-angle approach also required two signals for each corrective movement.)

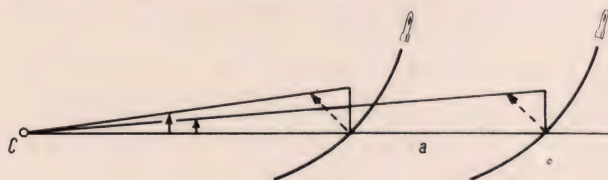


Fig. 11. Variable angular acceleration in a line-of-sight course  
 $a$  = Line of sight,  $C$  = Command post,  
 $\longrightarrow$  angular acceleration,  $\longrightarrow$  lateral acceleration.

The linear transverse acceleration of the bomb  $a$  is proportional to the stick deflection. The bomb-aimer, however, observing not the linear acceleration but — in relation to his point of view — the angular acceleration (Fig. 11), has the optical impression that, with increasing distance of the bomb, the effect of his stick deflection becomes less and less.

Moreover, the trajectory of the bomb intersects the line of sight under a continuously changing angle. Thus, the bomb-aimer will only perceive the partial effect of the acceleration component perpendicular to the line of sight.

### 5.2.2. Peculiarities of the Homing System

The difficulties of the acceleration control when using the three-point-coincidence method, which become disturbing to the bomb-aimer, are eliminated when the bomb is guided along a pursuit track according to the image presented on a television screen. Whereas, when using the three-point-coincidence method the bomb-aimer had to observe bomb and target from a third point, orientation is now effected in the missile itself and not influenced by the location

of the control stand. When looking at the television screen, the bomb-aimer is under the impression that he is moving with the bomb himself.

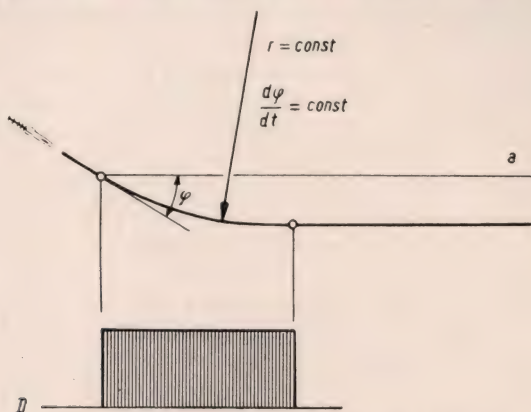


Fig. 12. Proportional angular velocity control  
 $a$  = Direction to the target,  $D$  = Joystick deflection.

A control deflection calling for a certain transverse acceleration forces the missile to pursue a circular curve as long as the stick deflection is maintained constant (Fig. 12). In relation to a space-fixed coordinate system, the missile will turn with a constant angular velocity around its lateral axis. Since, on the television screen, angles are represented by linear distances and angular velocities as linear velocities, a desired transverse acceleration, i.e. a certain control stick deflection, will correspond to an exactly defined linear velocity of movement of the target image across the television screen. This proportionality of the stick deflection and visual transverse velocity of the target image is maintained throughout the whole guided flight regardless of the changing position of the bomb in relation to the target or to the control stand. This proves to be a great advantage when compared with the three-point-coincidence method which did not show such proportionality between observed motion and control stick deflection.

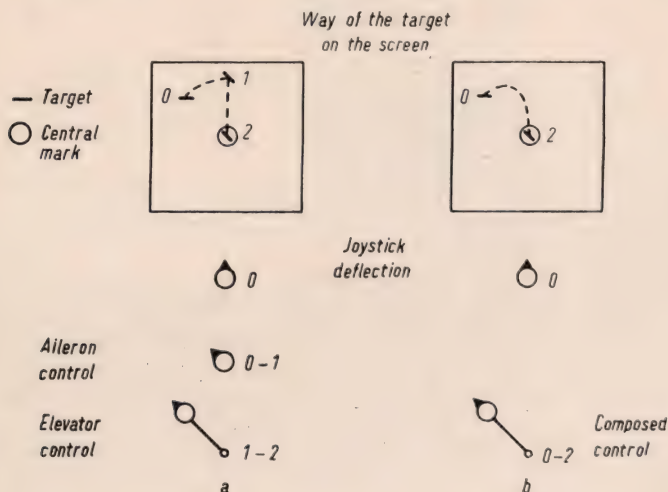
A control command affecting the elevator will, after a short delay which depends upon the response characteristics of the system, result in a proportional transverse movement of the target on the screen. As soon as the target image has reached the desired position, it is only necessary to return the control stick to its zero position and in doing so to take account of the response lag in the system. This offers another advantage over the three-point-coincidence method, in which a counter-command had to be given to cancel a previously given command.

### 5.3. Requirements of the Television System on the Bomb-Aimer

Whilst those characteristics of the television system discussed so far clearly facilitate the guiding control, another peculiarity of the television image was at first considered to be somewhat disturbing. Each time a roll correction was applied and the camera rotated together with the bomb around its longitudinal axis, the image on the screen rotated correspondingly, but in the opposite

direction. This complied of course with what the bomb-aimer would see if he were moving with the bomb. It appeared doubtful at first if the bomb-aimer could adapt himself to this phenomenon; later experience, however, showed that no difficulties arose in this respect.

Since the angle of attack was compensated by the wind vane displacing the objective lens, the image rotated around the optical axis of the camera aligned to the flight path. The optical axis was marked on the photocathode of the image storing tube by a little wire ring. This circular mark was transmitted as a part of the image and appeared in the centre of the screen (Fig. 13).



*Fig. 13. Effect of control on the screen*

When the target appeared on the screen in an arbitrary position to the centre mark (0 in Fig. 13), the bomb-aimer first rotated the control stick in order to rotate the image by rolling the bomb, thereby placing the target image in a position vertically above (or below) the centre mark (position 1 in Fig. 13). The next command given was a radial stick deflection which displaced the elevator and caused the target image to move with a corresponding velocity towards the centre mark (movement 1—2 in Fig. 13). Shortly before the target reached the centre mark, the control stick had to be placed back to the zero position.

It proved unnecessary, however, to give the two control commands separately, i.e. one following the other. Simultaneous control of roll and of pitch proved to be advisable in practice. When the target image was to be moved from its initial position to coincide with the centre mark, elevator and aileron controls were operated simultaneously by moving the stick in the direction of the first observed target deviation. Due to the lag in the control action, the target image moved then towards the centre mark along the trace 0—2, whilst the control stick remained in a fixed position. Experience gained with a simulator and with actual release tests showed that this combined control action was efficient once the bomb-aimer was sufficiently trained and had become accustomed to the television image.

One property of the television image has been emphasized from the start as a particular advantage: with the bomb approaching the target the image of the target would become larger and more distinct, so that aiming at a particularly vulnerable spot would become feasible. Objections were raised at first, however, that the bomb-aimer might become irritated by the target image seeming to explode just prior to the hit. This apparent explosion was expected to occur, due to the fact that the optical system chosen had an image-angle of  $13^\circ$ , in the last second prior to the impact.

Under unfavourable conditions of attack, however, the bomb at this stage would have already left the ideal pursuit curve and would be following its path of minimum radius, with no possibility left for correction anyhow. With a more favourable pursuit curve having a smaller radius of curvature, it would also during the last second be sufficient to hold the control stick rigidly in the same position as it was held immediately before, provided that the target image had so far remained in coincidence with the centre mark. It was found that this difficulty could be mastered if the bomb-aimer guided the bomb so as to avoid any large deviations for some time prior to the impact and remained calm until the impact actually occurred.

#### 5.4. Training of the Bomb-Aimer

##### 5.4.1. HFW (*Henschel Flugzeugwerke*) Simulator

A simulator was constructed by HFW to reproduce the subjective impression presented by the television screen during the guiding manoeuvre, and to have an opportunity for training the bomb-aimer (Fig. 14). The instrument was of a box-like shape, about  $40 \times 40 \times 60$  cm in size, with the control aids, including the control stick, at the front. A horizontally-mounted diapositive slide, depicting a target, was projected by a suitable optical system on to a screen mounted in the front panel above the controls.

The function of this device was to reproduce, in response to the movements of the control stick, the following image-movements on the screen:

a) For roll control signals:

Rotation of the image around the marked centre on the screen.

b) For pitch control signals:

Vertical shift of the image on the screen corresponding to the relevant stick deflection.

This task was solved — as far as can be remembered — approximately as follows:

A roll-command given by rotating the stick  $f$  caused the turn-table  $c$  to be rotated by the motor  $m_3$ , which together with the potentiometer  $e$  and a potentiometer connected to the control stick, formed a bridge circuit.

On the turn-table were arranged two perpendicular slides  $a_1$  and  $a_2$ , on which a diapositive frame  $g$  could be moved with motors  $m_1$  and  $m_2$ . With a sin-cos potential divider  $d$  mounted on the turntable-shaft and simultaneously feeding the two motors  $m_1$  and  $m_2$ , the resultant shift of the slide  $g$  was made always to appear as a vertical shift of the target image on the projection screen  $h$ .

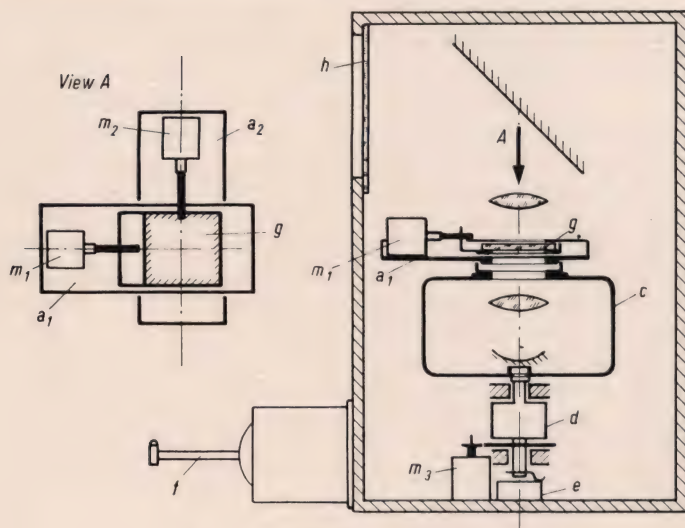


Fig. 14. Simulator for TV-guidance

The circuitry of the control set and motors took into account response characteristics of the missile, so that the movements of the image on the screen following a control manipulation corresponded to the actual movements of the target.

After becoming accustomed to the movements of the image on the screen and following some practise, it was quite easy to make the target coincide with the marked centre on the screen.

With this simulator, however, the progressive growth of the apparent target size and the subjective impression of the exploding target in the last phase prior to the impact could not be realized.

#### 5.4.2. Simulator Developed by the DFS (Deutsche Forschungsanstalt für Segelflug)

The DFS at Aining constructed a model based upon theoretical investigations regarding suitable flight paths and relevant guiding methods (see Dr. FISCHÉL's paper in this book, page 29, Fig. 4). It consisted of a target-carriage representing the target and its movements, a bomb-carriage, and a control desk from which the target-carriage was controlled. So as to avoid the complication of transmitting a television image to the control stand, the bomb-aimer was placed right on the bomb carriage. Distances, speeds and accelerations were reduced to a 1:400 scale, yet maintaining a 1:1 time scale. With this model, bombing runs from initial distances up to 4 km could be imitated within the correct flight time, whilst changes in movements of the target as well as other difficulties could be arbitrarily fed in from the control desk.

The growth of the target image as seen by the bomb-aimer was also truly imitated as long as the missile had a real distance of 40 m from the target<sup>8</sup>.

Thorough experiments with this model in which the missile flight-path curves discussed theoretically above and the appropriate guiding methods were

reproduced, fully confirmed the theoretical results. They proved the particular simplicity of the television-aided control along with a flight-path-fixed camera axis as chosen for the Hs 293 D; mastering the relevant control technique needed but a short training.

The model, primarily built for studying the various control methods, offered also the possibility for training bomb-aimers. Since, however, this plant required a lot of space and was difficult to move, a desk-model was developed by the DFS. This set was to have the same performance as the large plant <sup>7</sup>.

## 6. TESTS AND EXPERIMENTS

### 6.1. Preliminary Tests with the Television Equipment

In the course of development several series of tests were carried out, of which only a few dealing with the transmission, the image reception and the conditions within the missile will be mentioned.

#### 6.1.1. *Transmission Tests*

Previously, at a very early stage of development, systematic transmission tests had been carried out with the television equipment. They served to determine the possible transmission distances and gave the basic data for the choice of the transmission frequency.

Transmission tests between two vessels using a wavelength of 10 m resulted in transmission distances up to 40 km without disturbing reflections such as might be caused by heavy seas. At shorter wavelengths, however, such disturbing reflections were to be feared.

Transmission tests between aircraft were performed using a wavelength of 70 cm and a radiating power of 10 W. By transmitting a test image, distances up to 150 km were reached. Since both aircraft, if flying sufficiently high, were always in sight of each other, the short wave length was considered particularly suitable, especially since the aerial dimensions could be reduced on the aircraft and on the missile. Reflection disturbances, likely to occur with such short wavelengths, could be eliminated by the use of directed aerials <sup>2, 3</sup>.

#### 6.1.2. *Contrast Tests with Different Iconoscopes*

Regarding reception, tests were arranged to determine the discernibility of ground and sea-targets under various atmospheric conditions with respect to brightness and haze.

From a comparison of several iconoscopes with a varied sensibility for blue and for red it can be stated that in general, the red-sensitive iconoscope is superior, particularly in the presence of haze. On the other hand, for picking up airborne targets from below against the sky, the blue-sensitive was the more suitable one <sup>9, 10</sup>.

#### 6.1.3. *Operational Tests with Television Equipment and Control Apparatus*

Another test series was concerned with the joint operation of television and control equipment within the missile. The insensibility of the equipment to

shock and vibration had been proved already on delivery of the components; therefore no difficulties due to mechanical shocks caused by the hard action of the ailerons were encountered.

A further test investigated the possibility of feeding the television equipment and the control set from the same battery. Since the camera was fed by a rotary converter which had enough inertia to overcome sudden voltage drops such as those caused by the aileron-actuating magnets, and since the most important voltages were stabilized, no disturbances with such an arrangement were to be expected.

## 6.2. Testing of the Television Bomb

Flight tests with the Hs 293 D were carried out at the LUFTWAFFEN-ERPROBUNGSSTELLE at Peenemünde-West. During a series of flights without releasing the bomb, the behaviour of the whole equipment was checked under actual flight conditions. It was also checked that the auxiliary sight of the pilot was aligned to the optical axis of the television camera.

During the subsequent release tests the carrier aircraft first moved towards the target. Once the pilot had covered the target with his line of sight, i. e. when the target appeared also in the marked centre of the television screen, the bomb-aimer released the bomb. Following a short rocking movement caused by oscillation of the wind vane during the release, the image became steady and the bomb-aimer could guide the missile on to the target by manipulating the controls as described before.

The test drops were carried out on stationary targets only, mainly on a wreck lying off the coast. Only for a few tests was a marked ground area used as a target. Each test drop was recorded and evaluated by a cine-theodolite arrangement tracing the bomb path which, in the ideal case, should approximate a straight line. In addition, the television image was observed and, in some cases, also filmed by a second receiver so that the control procedure could be checked.

But since reports of these test drops no longer exist, apart from a few films, it is somewhat difficult to recall reliable data.

The target was well discernible on the image screen, though a few disturbances still crept in.

Loss of control of the bomb due to failure of the television system seldom occurred.

The control technique was easily mastered and resulted in satisfactory hitting accuracy.

Release tests were performed with about 70 bombs.

## 7. TREND OF FURTHER DEVELOPMENT

Having discussed so far the development stage of the television-equipped flying bomb as it was tested at that time, it may now be interesting to look at the plans for future development which, however, were not carried so far as to result in new equipment being turned out.

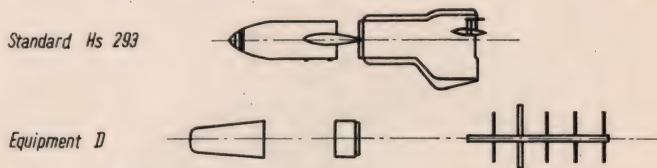
### 7.1. Preparations for Mass Production

Plans were prepared for mass production incorporating improvements of some details based upon the test results.

The FERNSEH-GMBH had already prepared a dual DOVE prism intended to replace the transverse movement of the camera lens operated by the wind vane. The advantage of this arrangement consisted in increased mechanical simplicity and in the opportunity of giving the camera hood at the forward end of the bomb a more favourable aerodynamic shape. Since with this arrangement the heated clear-sight screen, serving as front lid of the camera housing, could be omitted, the dual DOVE prism was provided with a heating element to prevent it from icing.

As an experiment, one camera was fitted with an azimuth-gyro, the position of which was transferred by a marker light and mirrors to the photocathode. In the first phase after the release, the bomb-aimer could use the light mark as an auxiliary target. As the launching of the bomb was performed in the direction of the target, the light mark on the television screen showed the target azimuth at the instant of release. Thus the bomb-aimer could guide the bomb on a straight course with the aid of this mark even when for some reason the target did not appear clearly on the screen. This procedure promised the following advantages:

- a) It was not necessary to wait, prior to the release, until the target was distinctly visible on the screen. The bomb could be dropped at the extreme distance of visibility, as soon as the pilot had covered the target with his auxiliary sight. Thus the possible release distance was increased to the maximum distance of visibility.
- b) Temporary fading of the television image in the initial stage of the pursuit curve could be bridged over by using the auxiliary target mark. This permitted bombing even when the sky was partially cloudy; the missile could be guided through layers of clouds in spite of the fact that no target was visible.
- c) Since the auxiliary target could be used with each release, special measures for initially caging the wind vane or for adjusting the airstream at the bomb suspension became unnecessary.



*Fig. 15. Hs 293 D. Design for production*

Mass production was envisaged for the following assembly units rather than for the complete television bomb (Fig. 15):

- a) camera section,
- b) fuselage centre section with the television transmitter,
- c) YAGI aerial.

The design provided for each Hs 293 to be converted by these assembly units into an Hs 293 D.

## 7.2. Improvement of Guiding Method

The advantages of the inverse rotation system have already been explained when discussing the various guiding characteristics. With this system, the missile path can be made to approximate a straight line after an initial phase of curved flight.

There already existed a gyro-equipped sight which worked according to this theory. It had been developed as an additional component for the LOTFE 7d and 8, in order to eliminate the influence of target and wind velocity vectors on this bomb sight. The equipment consisted of a reflex sight (REVI) which was deflected from the longitudinal axis of the aircraft by an azimuth gyro. The deflection angle between the line of sight and the gyro axis bore a fixed relation to the deflection angle between the aircraft longitudinal axis and the gyro axis. The gyro axis was held until the bombing run had covered the target with the line of sight, and then released.

The same technical means seemed suitable for attaining, under the conditions of television-aided bombing, a collision-course characteristic for the missile path. The one-dimensional angular deflection system of the above mentioned bomb-sight, however, had to be modified to become two-dimensional for this purpose. Thus, an azimuth gyro had to be fitted to the camera which was to deflect not a sight, but in this instance the optical axis of the camera itself by a corresponding angle. The following mechanical solutions offered themselves:

- Linear objective lens displacement, cross-fed by the gyro in two co-ordinates,
- Rotary displacement of two crossed dual DOVE prisms by the gyro.

With these arrangements, the lead angle of the missile path would have been attained by a deflection of the optical axis. The subjective impression on the bomb-aimer and his task would not have differed from the former system. The bomb-aimer would have to handle the controls as usual to establish coincidence of target image and centre mark on his television screen.

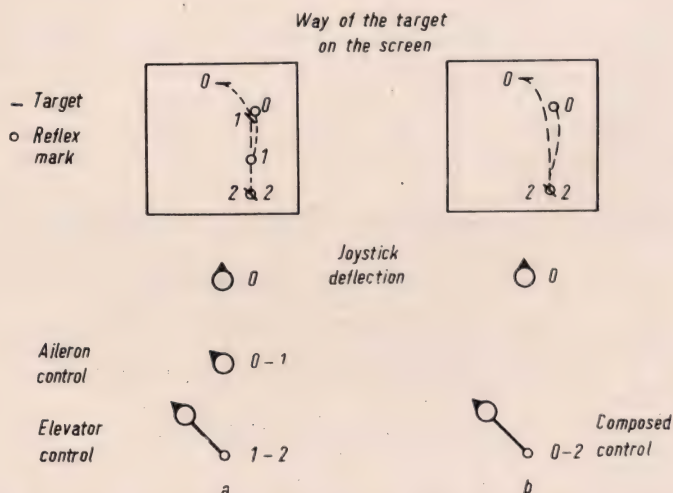


Fig. 16. Effect of control on the screen with inverse rotation system

This solution presented some technical difficulties, since either the gyro had to deliver considerable actuating forces or a servomechanism had to be installed. It therefore appeared more advisable to retain the position of the optical axis and to use a mirror system imposing a gyro-controlled light mark on the optical image instead. With this solution, the gyro was fitted with a mirror and had not to deliver any actuating forces. The optical axis remaining fixed to structure, each roll manoeuvre of the bomb caused the gyro mark to rotate together with the whole image on the screen. The position of the gyro mark in relation to the centre of the screen indicated the established lead-angle. The modified task of the bomb-aimer consisted now of obtaining, and holding, coincidence of the target image with a shifting reference mark instead of with a fixed one. Fig. 16 shows the relevant control procedure corresponding to the method outlined in Fig. 13.

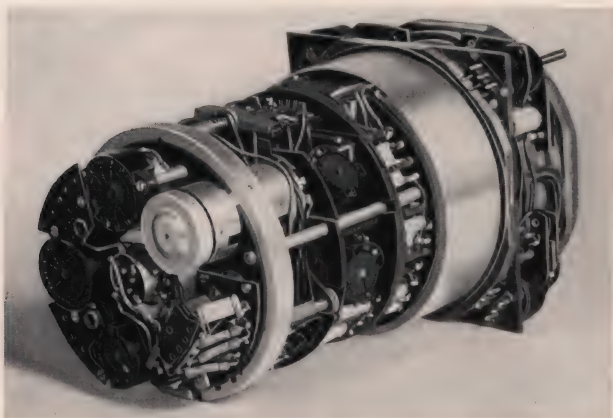
Here it should be stressed also that this gyro-controlled light mark results in the advantages previously mentioned, namely:

- increase of release distance, and
- possibility of releasing (and guiding) through clouds.

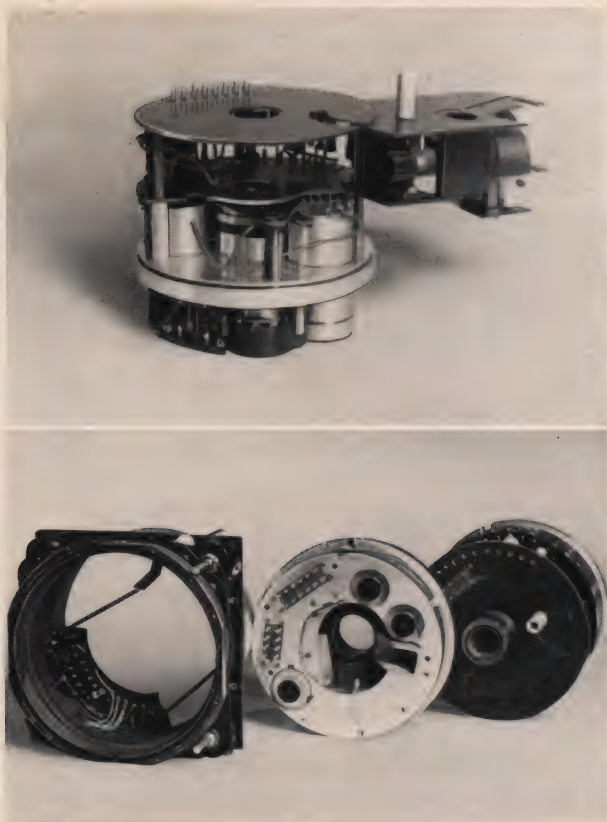
### 7.3. Further Developments of the Television Technique

Whilst the Hs 293 was tested with the television system described before, the FERNSEH-GMBH went on developing new equipment. The "TONNE" camera was sufficiently altitude-proof because of the rather low anode voltage of its image storing tube. The same could not be said, however, of the "SEEDORF" receiver, since it needed an anode voltage of about 6 kV. For high altitude conditions, the FERNSEH-GMBH developed a high performance receiver enclosed in an airtight housing (Fig. 17). The picture tube gave, with an anode voltage of 12 kV, a still better image. The receiver, skilfully composed of several partial chassis, was remarkable for its extreme compactness (Fig. 18). Heat dissipation from the tubes to the housing was effected by massive aluminium disks<sup>3, 4</sup>.

Also on the camera side further developments were under way. In collaboration with the TELEFUNKEN Corporation and with the Research

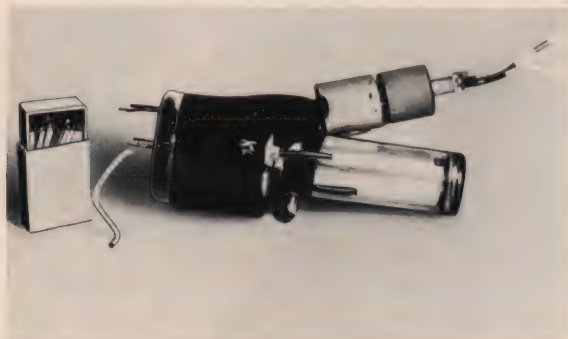


*Fig. 17. High performance TV-receiver*



*Fig. 18. Details of the high-performance TV-receiver*

Laboratory of the REICHSPST, the FERNSEH-GMBH developed the "SPROTTE" camera which contained a miniature image storage tube (Fig. 19) with a reduced number of lines resulting in smaller overall dimensions. The "SPROTTE" was mainly developed for anti-aircraft rockets, but did not reach actual test stage<sup>1</sup>.



*Fig. 19. Miniature iconoscope*

Another development started by the FERNSEH-GMBH was the "FB 50" with still less weight of equipment. The scanning was performed with 50 lines and a picture frequency of 25 c/sec. This type is worth mentioning for the extremely small frequency bandwidth of its picture transmission. Since with television guiding, jamming of the television transmission had to be reckoned with, the development of the "FB 50" might have offered the possibility due to its narrow bandwidth of transmission by wire. This development, however, also did not reach testing stage<sup>1</sup>.

## 8. SUMMARY

The development of the Hs 293 D which, until 1945, was the only television-guided missile being actually tested, made clear the fundamental peculiarities of this guiding method regarding tactical aspects, technical realization and requirements imposed upon the bomb-aimer.

More tactical freedom was attained by the fact that the carrier aircraft did not have to remain in the vicinity of missile and target after the release of the bomb.

The development of television equipment for installation in a missile had fully complied with the tactical requirements considering the possibilities offered by television techniques at that time.

Providing a homing procedure in connection with the already existing guiding control offered the much easier realized impression of a proportional turning rate control to the bomb-aimer.

Finally, it was found that the bomb-aimer could, without great difficulty, adapt himself to the television image and to the different relevant actions required from him.

In summing up, it can be stated that television-aided guiding of missiles had, under the then existing conditions, already attained promising efficiency.

## 9. ACKNOWLEDGMENTS

The author is indebted to Dr. R. MÖLLER, director of the FERNSEH-GMBH, and his collaborators for providing him with valuable informations and a series of illustrations about the television development.

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## DISCUSSION

Dipl.-Ing. K. WEMHEUER (Darmstadt): I would like to add the following remarks to the lecture delivered by Mr. MÜNSTER:

### 1. "TONNE" Television-Camera

The "TONNE" television-camera, though built-up as a very compact set, proved to be able to withstand influences of temperature and weather surprisingly well. One of the initial sets, mounted to the fuselage of the Hs 293 and protected only by the not at all airtight aerodynamic covering, was exposed to the Baltic autumn climate of the coastal lowlands. The set was installed in a tent measuring 10 by 12 m and was about one metre away from the ground. For a fortnight, no image disturbances were observed when switching-on the set every morning, though the air-moisture within the tent amounted to 100%; only after that period did creeping discharges cause disturbances which, however, disappeared after the set had been heated with dry air. — Another set was operated for several hours without trouble in a room warmed up to almost 40 °C by hot air.

### 2. Development Pursued by the Firm of LOEWE OPTA

Whilst the FERNSEH-GMBH developed the "TONNE-SEEDORF" set, the firm of LOEWE OPTA at Berlin-Steglitz constructed the "FALKE" television-set, in which the camera, though using the supericonoscope I S 9 of the FERNSEH-GMBH, scanned the image-field in spiral lines, which resulted in almost equal average resolution. This development led to a serviceable working-model. Since,

however, the test-series "TONNE" was about to be delivered, the development "FALKE" was dropped following the general reduction of development programmes. Instead, the firm of LOEWE OPTA was proposed for the construction of operational testing-sets for the "TONNE-SEEDORF" equipment, but this intention did not pass the stage of initial informative discussions.

The spiral scanning with constant angular velocity was well suited for target locating purposes, since the image on the receiver screen showed a higher resolution in the central part of the image. The center spot on the screen did not disturb the image-impression as a whole. The camera "FALKE" required considerably fewer valves than the "TONNE" camera.

### 3. Development Pursued by the Firm of GOLLNOW & SOHN

The firm of GOLLNOW & SOHN, Stettin, developed a bomb which was to guide itself on to a ship-target according to the picture received by a built-in television-camera. With spiral scanning, corresponding control commands could be derived quite easily from the signals of the four image-quadrants, either for the direction of view of the television-camera prior to the release of the bomb, or for the guidance of the bomb after release. Auxiliary circuits, e.g. those for scanning image-parts outside the target diameter, were feasible, by means of which deception-manoevres of the target — e.g the firing of flare lights — could be rendered ineffective. A prototype of the television set using spiral scanning with about 100 to 200 spiral lines was developed by the FERNSEH-GMBH on behalf of the firm of GOLLNOW. Another model with improved dynamic characteristics regarding the adjustment of the optical axis was built by the GOLLNOW firm itself. This set worked satisfactorily in 1942 in a sea- and ship-model test under conditions of light approaching reality as far as possible. The rapid oscillations of the body-axis possible during flight were not, however, sufficiently considered in this model. This development was also dropped in favour of the "TONNE" equipment.

# SUMMARY OF THE DEVELOPMENT OF HIGH-FREQUENCY HOMING DEVICES

GEORG GÜLLNER \*

## INTRODUCTION

It was only during the second part of World War II that, with the aim of avoiding the well-known disadvantages of the optical, infra-red and acoustical homing devices, the development of seven various types of high frequency homing devices was begun. By that time, it should have been clear that these devices which were quite expensive, would probably be too late to go into action. The Research Laboratories involved were at the BLAUPUNKT plants, and at the REICHSPPOST Research Institute (REICHSPPOSTFORSCHUNGSANSTALT). The homing devices were designed to operate on either the active, the semi-active or the passive principles. The active and semi-active principles worked as proper reflection devices, which emitted radio frequencies from transmitters located in missiles or on the ground, and used the reflected energy for their navigation. The passive principle was designed for use against targets which themselves emitted high-frequency energy.

As far as can be recalled, the following devices were under construction:

BLAUPUNKT:

1. "MAX A",  
operating on the active principle on 3.9 cm wavelength = 7700 Mc/sec, with an output of 5 Watts of uninterrupted r.f.
2. "MAX P",  
a passive device, which was to locate the radar sets of aircraft which worked on the 3 cm = 10,000 Mc/sec band.

REICHSPPOST Research Institute (REICHSPPOSTFORSCHUNGSANSTALT):

3. "RADIESCHEN",  
passive device for short- and ultra-short-waves. It was to be used against ground transmitters.
4. "WINDHUND",  
a passive device similar to "MAX P".
5. "DACKEL",  
this was an active device and operated by impulse system on 70 cm = 430 Mc/sec.
6. "MORITZ",  
a device working on the semi-active principle on 70 cm = 430 Mc/sec.

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## 1. "MAX A"

The particulars of this device are relatively well known. It was to operate on the active principle on  $3.9 \text{ cm} = 7000 \text{ Mc/sec}$  with uninterrupted r.f. and made use of the DOPPLER effect. The energy, which was emitted by the transmitter antenna and reflected by the target, was picked up by two array systems, one for the horizontal, and one for the vertical vector. It was fed, through high frequency and low frequency switches to a differential steering relay (Fig. 1). This relay made use of the SIEMENS navigation system, and transferred the

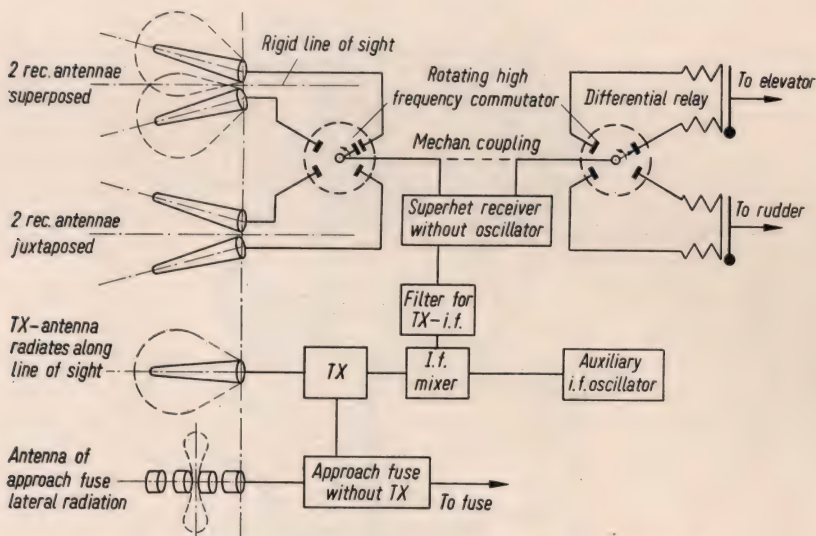


Fig. 1. Working principle of "Max A" and "Max P" devices

course correction impulses to the control system of the missile (Fig. 2 and 3). The voltage difference of the array systems, which corresponds to the deviation of the target, was used to correct the course until the target lay in the plane of symmetry of both array systems. The comparison of the voltages from the arrays can be done either by using two absolutely similar receivers, or by switching the arrays on one receiver. But the use of two or four receivers (for all four co-ordinates) is very expensive, and, because of the rapidly changing field intensities it is very difficult to build receivers with identical amplification characteristics. These were the reasons which led to the idea of switching the four antennas, one after the other, on one receiver.

For the correct navigation of the missile, the following requirements had to be met:

- precision of adjustment:  $\pm 1^\circ$ ,
- maximum distance of missile from target: 30 m,
- angle of action:  $\pm 10^\circ$ ,
- reaction speed:  $30^\circ$  per second,
- range distance: at least 1000 m,
- correction voltage as function of deviation of target: linear up to  $\pm 5^\circ$ .

Because of the size of the missiles, the dimensions of the transmitter were designed to be as small as possible.

### 1.1. Transmitter and Receiver

The transmitter was operated on a frequency of approximately 7700 Mc/sec and emitted approximately 5 Watts of energy from a horn antenna (Figs. 3 and 4). The transmitter tube was an 8-slot-magnetron working with a plate voltage of 600 Volts and a field intensity of 1000 Gauss. A certain amount of

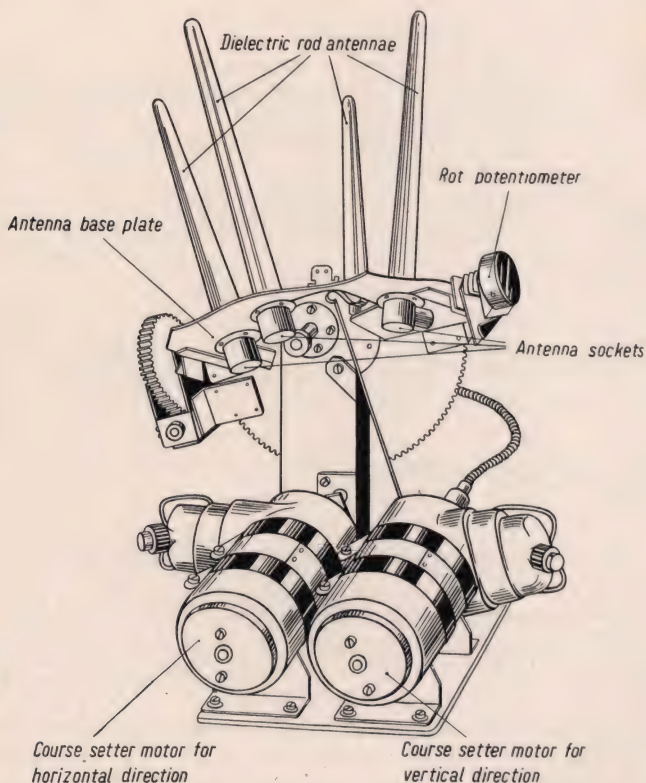


Fig. 2. Dielectric rod antennae and control system of "Max A" and "Max P" devices

energy was tapped from the transmitter and mixed with an oscillator frequency to obtain an intermediate frequency of 300 Mc/sec. In this way two oscillator frequencies of 8000 Mc/sec and 7400 Mc/sec respectively were produced. The 8000 Mc/sec frequency was picked out by a filter and used as an oscillator frequency for the intermediate frequency amplifier. In this way a rigid relation was maintained between the transmitter and oscillator frequencies, and it was possible to use a narrow, high amplification intermediate frequency band.

Depending on the relative speed the energy reflected by the target varied in frequency, due to the DOPPLER effect, from 4000 to 7000 c/sec from the original

transmitter-frequency. A super-regenerative amplifier with a bandwidth of approximately 1 Mc/sec on 300 Mc/sec amplified both incoming frequencies. In addition to this setup, an amplifier regulator was provided which had a time delay device designed to prevent operation whilst the antennae were being

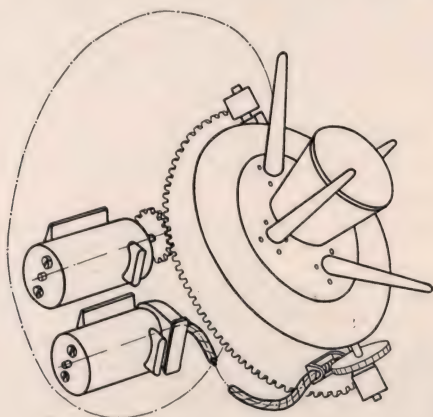


Fig. 3. Antenna disc with transmitting horn and receiving dielectric rods

switched but which enabled field intensity variations to be corrected during the approach to the target. After rectification, the low frequency was passed through low-frequency filters and again rectified and then fed into the navigation system. The sensitivity of the receiver is said to have been  $100 \text{ KT}_0$ .

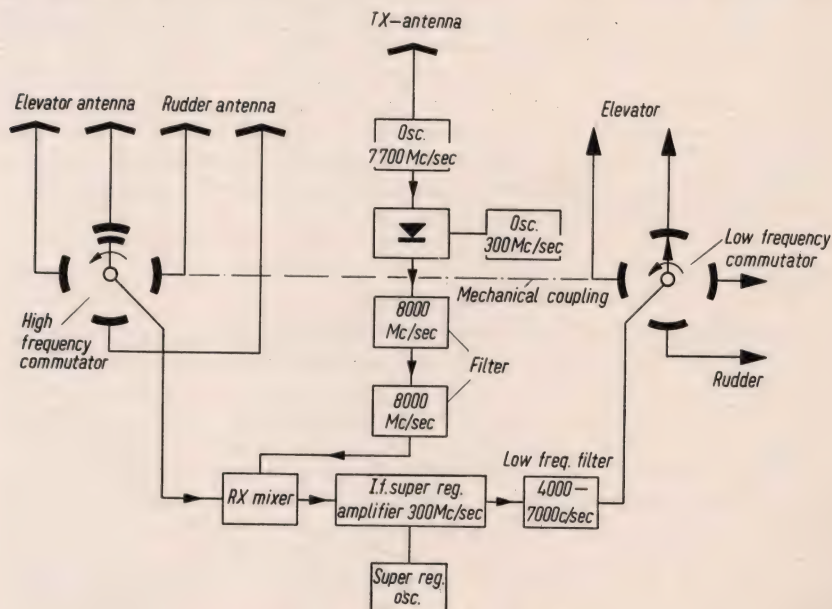


Fig. 4. Block diagram of "Max A"

## 1.2. The Antenna System

The receiver antennae were built as di-electrical rod beams with a half-lobe angle of  $\pm 15^\circ$ . In the case of a  $1^\circ$  displacement of the target, the deviation of the receiver antenna from its  $15^\circ$  mean position produced sufficient change in receiver output to operate the differential relays and to change the course of the missile. In order to produce better decoupling, the transmitter antenna was constructed as a horn beam of sheet metal. When properly set up, a power decoupling factor of  $1:10^5$  was obtained.

Two array head systems were constructed:

1. A rigid antenna head, fixed in the axis of the missile. This head was designed for pursuit courses;
2. A movable antenna head for collision courses. Its range was larger than that of 1.

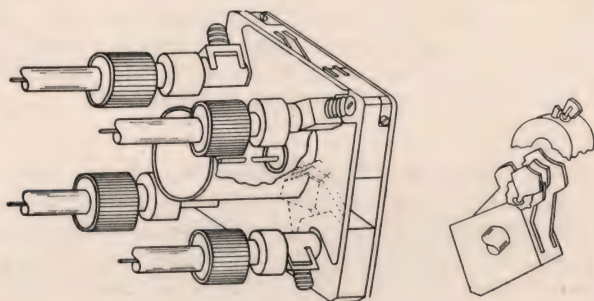
The pursuit course system was to be used in air-to-air operations, whereas the collision course was planned for ground-to-air use. When operating with the collision course system it was first necessary to guide the missile from the ground, until it came within range of its own device. After locating the target, the target-following-head in the first stage of the array automatically turned itself towards the target without affecting the navigation of the missile. After reaching a distance of approximately 1000 m from the target, the angle between the axes of the missile and the antenna head was fixed and automatic navigation was begun with the aid of a potentiometer. The head turned in the direction of the target, moved the potentiometer, and upset the balance of a resistance bridge which altered the course of the missile until the fixed angle between the array head and the missile axis was again obtained. The control impulses depended on the fact that when setting the angle between missile and array the whole potentiometer was moved, whereas the automatic navigation correction depended only on the movement of the slide wire of this potentiometer. The switching was done by a relay. The receiver antennae were fixed, corresponding to the co-ordinates, to a cruciform plate, with the horn beam array in the centre. The movement of the antennae were produced by two course-setter motors, one of them moving the antenna disc directly, the other one producing the rectangular movement through a flexible shaft. It was therefore possible to set the antenna to any direction in space (Fig. 3).

## 1.3. The Switching System

Great difficulties were experienced with the switching systems and a proper solution was not obtained. Due to the monotonic variation of the field intensity, the switching speed had to be high enough to prevent unwanted deviations of the missile whilst the antennae voltages were being switched. At a target distance of 500 m and a relative speed of 100 m/sec, with a switching frequency of 50 c/sec, the deviation was approximately  $\pm 0.2\%$ . It was unsuitable to have the switching system operate too fast, because it was necessary to let the switching transients fade away, and equally necessary to avoid a collision between switching frequency and the DOPPLER frequency. The original construction of these systems was based on continuous commutation. As a result, the incoming signals were modulated by the 50 c/sec commutation frequency, and its harmonics, to such an extent that it was not possible to use

the system for control at its highest sensitivity. But short-circuiting the low frequency, whilst switching, also produced no results, because the commutation curve of the switch did not have an ideal rectangular form. To correct this difficulty, jerky movements were provided by a Maltesian cross. The switching was done in the following manner: → low frequency short circuit → reversal of antenna → release of the low-frequency short circuit → switch stopped to enable high frequency transfer → low-frequency short circuit → reversal of

It was found that proper timing of the switching reduced the commutation jamming to a tolerable degree. Fig. 5 is a picture of the high-frequency commutator. Special troubles arose from the adaptation and the equal transfer



*Fig. 5. High-frequency commutator connecting the four receiving antennae, one after the other, with the receiver*

of the different four antenna voltages. The transfer of the energy was done by capacity or by an E-wave tank circuit and the commutator was adapted by high-frequency iron screw to the four equal antenna voltages.

#### 1.4. The Approach Fuse

No detailed information on the approach fuses is now available. A certain amount of energy was tapped off from the transmitter tube and led to a dipole array with a lobe characteristic as shows Fig. 1. The array worked as a transmitter and receiver antenna simultaneously. The emitted energy was reflected by the target and returned via the same antenna to the detector of the receiver. When the missile was near enough to the target, the increasing reflected energy produced a minimum of the DOPPLER-beat frequency and operated to fuse, provided that the missile was perpendicular to the target.

#### 1.5. Results and Difficulties

Theory showed that, provided the transmitter output amounted to 5 Watts, the receiver sensitivity was  $100 \text{ KT}_0$ , and the reflecting surface was  $1 \text{ m}^2$ , a range distance of 2000 m could be obtained. Ground tests proved that this performance was possible. A great disadvantage to the accuracy of the direction finding was the percussion sensitivity of the whole device.

Another difficulty was to be expected from the fact that the field intensity of the reflection between target and missile does not show a monotonous curve.

This curve will show, in accordance with the size of the target, maxima and minima, possibly closely-spaced. This may influence the homing, and a solution to this problem was not found.

## 2. "MAX P"

The main constructional features of this system were similar to these of "MAX A" but because it had no transmitter, it weighed less and was less expensive. The range of this device in the 3 cm band proved to be very large and it is said to have attained 50 km against air-borne radar. The difficulties lay in the fact that the antenna of radar transmitters was directed on the missile every 2 to 3 sec, so that navigation impulses were only possible with time intervals of 2 to 3 sec. It was planned therefore, to use also the second lobe maxima of the transmitting antenna for navigation impulses. The semi-active system "MAXIMILIAN" was planned on this basis.

## 3. "RADIESCHEN"

This device operated on the passive principle on the short- and ultra-short wave bands, and in the first place was scheduled for use against ground transmitters (e.g. LORAN-transmitters). It used the fact that the direction of the electric field, and therefore the direction of the transmitter, was perpendicular to the electric and magnetic vector of the field of radiation. Therefore, the position of the transmitter could be found from an analysis of the electric and magnetic vectors. Initially the system worked on a wavelength of 3 m, it required 5 to 7 valves and had a weight of 15 kg. The direction analysis produced a direct current, which was used for navigation. The operation angle was very large and is claimed to have reached  $\pm 90^\circ$ . Distance tests, which showed results up to 80 km, do not mean much, because the test transmitter was very powerful and had an output of 100 kW. The development of this system got so far as the launching of six test vehicles and the tests, on which Dr. KLEINWÄCHTER will give a special report, had good initial results.

## 4. "WINDHUND"

This device worked on the passive principle and as in the case of "MAX P" was designed for use against aircraft radar sets on the 3 and 9 cm bands. It was to be used in high speed missiles so as to prevent the radar device from being switched off after detection of the missile, or at least to have it switched off too late. The disadvantage of this system was (similar to "MAX P"), the intermittent navigation impulses which, in combination with the high speed of the missile, prevented proper short range location.

## 5. "DACKEL"

"DACKEL" was an active device working on 70 cm (430 Mc/sec). It used the radar impulse system. The close-up detection was to reach 50 m, a distance, which probably would not have been attained. The device was very expensive.

## 6. "MORITZ"

"MORITZ" was designed to work on the semi-active principle. The transmitter was located on the ground and had a continuous output of 10 Watts. Its wavelength was 70 cm (430 Mc/sec). The receiver made, as did "MAX A", use of the beat frequencies due to the DOPPLER effect between the direct wave and the reflected wave, and navigated on the latter. The trouble of this system was the possibility of the missile locating and attacking the transmitter instead of the correct target. The advantages were thought to be

1. the small weight and the low costs, since the missile had no transmitter, and
2. that there were no difficulties in de-coupling the transmitter and receiver antennae.

This device also was still in the development stage.

## 7. CONCLUSION

All except one of the six devices described above had not yet reached the point where flying tests could have started. Mainly test work and laboratory construction had been carried out. There is no doubt that the completion of the work would have still taken a long time, as the real difficulties arise when flying tests start. Since the development had begun only a short time before the war was over, it was impossible to complete it before the end of the war.

## DISCUSSION

Dr.-Ing. KLEINWÄCHTER (Saint Louis, France): In his review on the development of high-frequency homing devices Dr. GÜLLNER had only devoted a couple of words to the electromagnetical homing device "RADIESCHEN". I would like to complete his quotation by a couple of remarks. I want to mention this homing device, which I had invented and developed together with less than a dozen engineers working on antennae in the department of Dr. BRÜCKMANN in the REICHSPOST-FORSCHUNGSANSTALT in 1943 to 1945, because it is, as far as I know, the one and only electromagnetic homing device which during World War II had reached the flight test stage. Apart from that, several technical developments pertaining to this problem may be of general interest.

The homing device "RADIESCHEN" was mainly scheduled for operation against the hyperbolic navigation transmitters located at the south and east coasts of England which guided bomber squadrons over Germany, with the aim of destroying them with self-navigation missiles.

The name "RADIESCHEN" derives from the combination of the German words for the two receiver devices, the frame (*Rahmen*) and the dipole (*Dipol*) thus giving the word "Radi", which means radish and is generally called "RADIESCHEN". The reception voltage at the frame indicates the position of the magnetic field vector; the voltage at the dipole indicates the position of the electrical field vector of the electromagnetic target transmitter. This gives also the direction of the POYNTING radiation vector  $\vec{S} = \vec{E} \times \vec{H}$ , which passes through the target at a distance. This homing system might be called the

polarisation system, since it does not react on the phase differences of waves, but on the polarisation direction of  $\vec{E}$  and  $\vec{H}$ . This means that it may also be used for waves which are very long compared to the dimensions of the aerial.

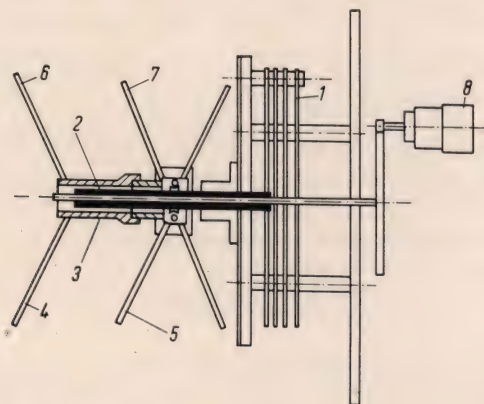


Fig. 1. Self-homing device "Radieschen"

1 = Loop, 2 = Fix dipole, 3 = Hull, 4 = Mobile dipole, 5 = Wobbler, 6 and 7 = Counterprise, 8 = Driving motor

Fig. 1 shows the navigation system with the rigid frame (1) and the dipole (2). The frame centre line and the dipole axis coincide with the axis  $A$  of the body. When  $A$  points in the direction of the POYNTING vector  $\vec{S}$ , neither the dipole nor the frame receives any voltage.

In order to give a virtual hunting movement to the dipole as well as to the frame, the dipole is connected to the eccentric dipole tag (4) by a metal cylinder (3), which works as a rotating capacitance coupling. Also the magnetic field of the frame is deviated by an inclined metal stagger disc (5) located in front of it. The rotation of the tag and of the stagger disc by means of the motor (8) and a non-conducting driving shaft through the dipole results in a conical movement of the effective dipole and of the effective frame directions around the axis of rotation  $A$ . This gives the desired hunting movement to the frame and the dipole by avoiding troublesome rotation couplings and uses the lightest possible rotating parts. The high frequency voltages of the rigid frame and of the dipole were amplified and demodulated and, after comparison of their phases with those of the rotating tag and the stagger disc in the usual manner, used as D. C.-impulses.

In order to obtain navigation impulses only from the angle deviations of  $\vec{A}$  relative to  $\vec{S}$  and not due to variations of the transmitter output, we gave the receiver a logarithmic sensitivity characteristic.

Fig. 2 shows how the homing device was mounted on the controlled free-falling bomb FRITZ X. The homing device "RADIESCHEN" was mounted in the tail for construction reasons. A minute study of field deformations due to the flying body revealed that this mounting system is admissible when the wavelength is large compared to the size of the flying body.

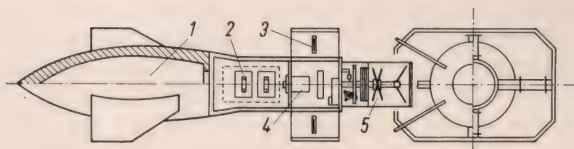


Fig. 2. Guided bomb Fritz X with self-homing device "Radieschen"  
1 = Explosive, 2 = Receiver, 3 = Spoiler, 4 = Gyroscope, 5 = "Radieschen"

The difficulty of being able to adjust, with a mounted "RADIESCHEN", only the body axis  $A$ , but not the unknown trajectory tangent during the flight to the target, was overcome by the following system: by setting the rudder or spoiler of a delta stabilized flying body to its zero position, the longitudinal axis adjusts itself, after decreasing oscillations, to the direction of the trajectory tangent. The "RADIESCHEN" navigated FRITZ X switched off the elevators and the rudder consecutively, so that it consecutively measured in one direction and navigated in the other. To adjust the axis of the flying body as fast as possible,

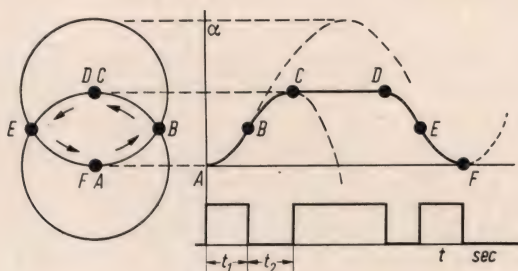


Fig. 3. Controlled adjustment

the zero and extreme position impulses were given by the programme shown in Fig. 3. To set the initial incidence angle  $\alpha = 0$  to the angle  $C$  resulting from the full opening of the spoiler, the spoiler was first opened for the time  $t_1$  seconds, then closed for  $t_2$  seconds, to be opened again after that and to remain so. In this way the axis of the flying body moved from position  $A$  into position  $C$  without passing through  $B$ . A similar navigation impulse is given, after switching off the spoiler, during the interval  $EF$ , so that the axis of the body moves into the weather-vane position  $F$  at zero incidence. The switching periods  $t_1$  and  $t_2$  depend on the oscillation of the body and, therefore, on the dropping time. They were, therefore, ordered by a cam switch which rotated according to a preset programme.

On the 23rd August 1944 the first "RADIESCHEN"-equipped FRITZ X was dropped by a He 111 at 7000 m on a 500-watts ground transmitter operating on 5 Mc/sec and situated on the test grounds of RHEINMETALL-BORSIG in Leba. The trajectory was controlled by three ground theodolite cameras and proved that the control system was working correctly throughout the trajectory. In addition the movements of the flying body were filmed from the He 111. We could clearly see how the side rudder made the body meander towards the

target. The two dropping tests with FRITZ X showed the side oscillation of the drop line to be less than  $\pm 10$  m, whereas in both cases the point of impact was estimated at 30 to 40 m from the target. This was due to the fact that the homing navigation was operating from the moment of the release and therefore that the flying body was pressed downwards during the upper horizontal trajectory. Accurate hitting in both directions would have only been possible after finding the proper release area and after a proper vertical trimming of the "RADIESCHEN".

Since from the summer of 1944 on flights to England with the carrier planes for FRITZ X, the He 111 and Do 217, were no longer possible, we switched over to flight experiments with lighter flying bodies. Until the end of the war ten B-246 gliding bombs of the firm of BLOHM & VOSS equipped with "RADIESCHEN" were launched on the artillery ranges at Unterlüss. Fig. 4 shows the B-246

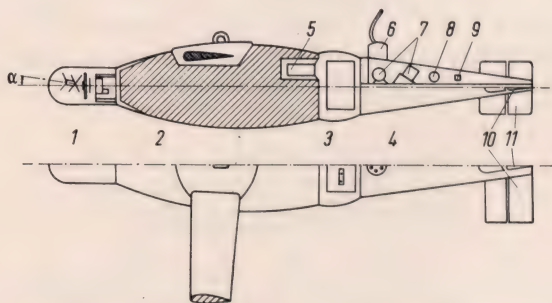


Fig. 4. Guided glide bomb BV 246

1 = Directional finding set, 2 = Tank for explosive, 3 = Receiver, 4 = Battery, 5 = Recorder, 6 = Connection, 7 = Gyroscopes, 8 = Transformer, 9 = Magnets, 10 = Elevator, 11 = Rudder

equipped with the homing device. Due to the advantageous aerodynamic form, this flying body had a gliding angle of 1 : 25, so that, if dropped from 10 000 m, it reached a distance of more than 200 km. The gyro course navigation of the B-246, which received the impulses from "RADIESCHEN", had not yet reached the stage of flight stability necessary, so that only two dropping tests did not result in a premature crash of the body. In these two cases, the homing device operating on the rudder controls drove the flying body only a couple of yards off the target transmitter.

# THE GUIDANCE SYSTEMS HV AND HAWAII II

RICHARD SCHÄFER \*

## 1. THE GUIDING OF A FLYING OBJECT IN A GUIDE BEAM

### 1.1. The Generation of a Guide Beam

The good results obtained at the beginning of the war with the "WÜRZBURG" radar equipment for the spacial location of aircraft, suggested that the field changes occurring in the narrow angular region around the axis of the antenna, if properly interpreted, could be used for guiding flying objects of any sort along this axis.

As is well-known, such a guide axis or guide beam is formed when the centre line, i.e., the direction of maximum energy of a revolving antenna diagram, rolls around a conical surface.

The simplest method of producing this type of movement is to rotate a primary radiator in a small circle around the focus of an axially-symmetric parabolic reflector. In this case, the primary radiator may be a half-wave dipole or any suitably shaped wave-guide opening. Depending on the type of radiator, the direction of polarisation of the electric field vector may also rotate, or it may remain parallel to a given direction. On smaller antennae, the rotation of the antenna lobe may be obtained by the reverse procedure, i.e., the primary radiator remains fixed and the reflector is moved in a suitable fashion.

### 1.2. The Field Modulation of a Guide-Beam Device

If we now consider any point in space surrounding the guide beam, then the field existing there appears to be modulated at the frequency of revolution of the diagram. The modulation-characteristic and the degree of modulation are determined by the shape of the directive diagram and the degree of deflection; the phase is determined by the angular position of the point of observation.

Fig. 1 shows a section of a plane passing through the guide beam and the point of observation, together with the rotating directive diagram.

In this the antenna characteristic was assumed to have a width at half-maximum value of  $5^\circ$  and its centre line is inclined at  $3^\circ$  to the axis of the guide-beam antenna. For an angular deflection of  $1.6^\circ$  the variation of the energy received was about 9 db, as it is assumed here, the characteristic was similar to that of the WÜRZBURG-RIESE radar equipment.

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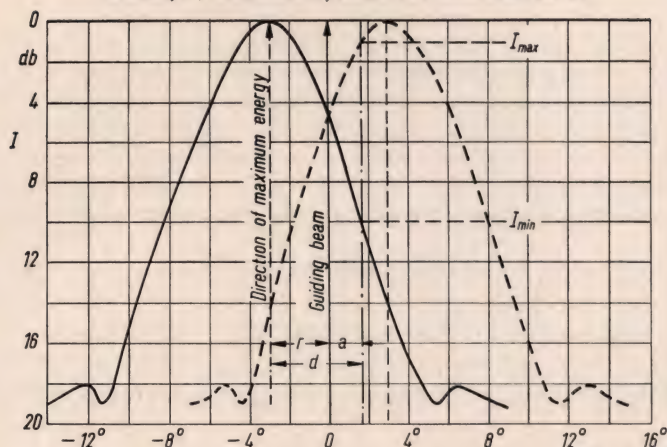


Fig. 1. Radiation pattern of a guiding beam antenna  
 $a$  = Deviation from guiding beam

Now, it is easily seen that if there is another frequency present, the phase of which, however, must be independent of the momentary position of the reference point, information can be obtained about the spacial position of the reference point, by measurement of its phase difference from the modulation characteristic. In principle, this auxiliary frequency could be generated within the flying object, and then compared with the demodulated frequency. However, the requirements of constancy of frequency and phase which would have to be met, could only be achieved on board of the flying object at the expense of considerable electronic complication.

Now the advantage of the HV method is that this reference frequency can be obtained directly from the modulation characteristic, and therefore any frequency or phase variations which arise will always be in the same direction for both frequencies, and are eliminated on formation of the phase difference.

A further important consequence is that roll-stabilisation of the flying object is unnecessary, and the latter may revolve in any way about its longitudinal axis, within the area of field distribution rotationally symmetrical to the guide beam, provided that its speed of revolution does not exceed a certain value.

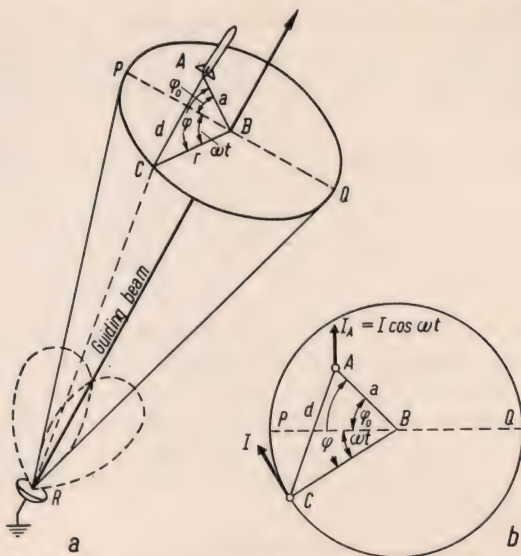
A good qualitative illustration of this may be obtained from a simplified mathematical approach.

Two important conditions must be satisfied to bring about HV guiding:

- The defocussed primary emitter for the production of the rotating directive diagram must be plane polarised, which can be achieved most effectively with a half-wave dipole.
- The receiving antenna of the flying object must also be plane polarised, and it is therefore expedient to give it too the form of a half-wave dipole.

Let the transmitting dipole now rotate with an angular velocity  $\omega$ , the direction of maximum energy tracing out a cone in space.

In Fig. 2a the flying object  $A$  is situated at a distance  $a$  from the guide beam  $B$ , and at any time  $t$ , it has a distance  $d$  from the instantaneous centre of the diagram  $C$ .



*Fig. 2.*

a) *Triangular co-ordinates in the plane perpendicular to the trajectory*

*b) Polarization of the transmitter and receiver dipoles at the time  $t$*

Assuming that the deflection  $a$  remains small, the problem may, in practice, be treated as a planar one, and we get

$$(1) \quad d^2 = r^2 + a^2 - 2ra \cos \varphi,$$

where  $\varphi = \omega t + \varphi_0$  is the angle of revolution of the rotating diagram.

Provided that the directive diagram itself possesses rotational symmetry about its centre line  $RC$ , and that in the vicinity of the guide beam  $d \approx r$  (see Fig. 1) the momentary guide planes form straight lines of section with the diagram, then the field intensity  $I$  can be represented approximately by the function

$$(2) \quad I = I_0 - s d$$

Here  $I_0$  is a suitably chosen constant, and  $s$  the average slope at the point of section.

As a first approximation from equation (1) gives

$$(3) \quad d = r \left( 1 - \frac{a}{r} \cos \varphi \right)$$

and this, together with (2) yields

$$(4) \quad I = I_0 - s r + s a \cos \varphi.$$

When  $d = r$ , the field intensity in the guide beam is given by equation (2) as

$$I_B = I_0 - s r$$

or from (4)

$$(5) \quad I = I_B + s a \cos (\omega t + \varphi_0).$$

Here  $\varphi_0$  is the angular deflection of the flying object from any chosen reference or guide plane, e.g., the plane through the horizontal and the guide beam.

Now it follows from equation (5) that at any point  $A$  in space there will always be a field, of intensity  $I$ , which is modulated at the rotational frequency of the transmitting dipole. The degree of modulation depends upon the slope of the antenna characteristic and the magnitude of the deflection  $a$  from the guide beam; the phase of the modulation gives the angular position of the flying object in the plane  $ABC$  (Fig. 2a) with reference to the guide plane  $PQR$ .

Without loss of generality, it may be further assumed that the receiving antenna of the flying object is polarised perpendicularly to this guide plane  $PQR$ . The relationships illustrated by Fig. 2b then exist.

Within a plane passing through the reference point  $A$  at right angles to the guide beam, the direction of polarisation of the field  $I$  will revolve with an angular velocity  $\omega$ . The intensity of the field received by the plane-polarised antenna at the point  $A$  is then

$$(6) \quad I_A = I \cos \omega t.$$

The intensity of the field entering the receiver  $A$  is given by equations (5) and (6) in terms of magnitude and phase as

$$(7) \quad I_A = f(t) = I_B \cos \omega t + s a (\cos^2 \omega t \cos \varphi_0 - \frac{1}{2} \sin 2 \omega t \sin \varphi_0).$$

In this, the origin of co-ordinates  $\varphi_0 = 0$  was chosen, as already stated, perpendicular to the direction of polarisation of the receiving dipole.

The horizontal deflections are given by putting  $\varphi_0 = 0$  and  $\varphi_0 = 180^\circ$ , and the vertical deflections by putting  $\varphi_0 = 90^\circ$  and  $\varphi_0 = 270^\circ$ .

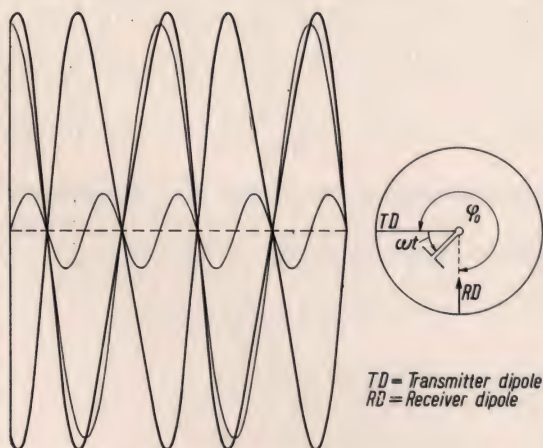


Fig. 3. Field modulation of vertical or altitude deviation

$$I_A = I_2 \cos \omega t + \frac{1}{2} s a \sin 2 \omega t \text{ at } \varphi_0 = 170^\circ$$

Naturally, the terms horizontal and vertical are only correct for an observer flying with the aggregate; for an observer on the ground it would be more appropriate to say "deflection perpendicular to the momentary direction of the receiving dipole" instead of horizontal, and "deflection in the direction of the receiving dipole" in place of vertical. However, as the terms "horizontal deflection" and "vertical deflection" came into general use during the war, they will also be retained for the following considerations.

In Fig. 3 as an example the modulation in the case of vertical deflection and  $\varphi_0 = 270^\circ$  is shown. In the case of a  $90^\circ$  rotation of the aggregate about its longitudinal axis "horizontal" and "vertical" change places, but since the rudders and elevators also rotate, the guiding signals are still given in the correct phase. Now if the flying object rotates at constant speed it will appear to the receiver that the angular velocity of the transmitting dipole has increased, or decreased, depending on the sense of rotation. Therefore, since frequency dependent terms must be used in the evaluation of the modulation characteristic, this autorotation will produce false signals unless, as is usually the case, the angular velocity is low compared with the speed of revolution of the dipole.

### 1.3. FOURIER-Development of the Modulation Characteristic

The modulation characteristic  $f(t)$  contains the double frequency of rotation of the transmitting dipole, and its phase is independent of the angular deflection from the guide beam. Therefore a fixed reference phase can be derived directly from it. The phase of the rotational frequency varies with the position of the point of observation.

For the purpose of computation of the magnitude of the amplitude and phase, it is best to express equation (7) as a FOURIER series. The low-frequency evaluation of the modulation characteristic is carried out after the second rectification and, for the sake of simplicity, the rectifier is assumed to be linear. From the field characteristic  $f(t)$ , the demodulated characteristic  $f^*(t)$  is obtained by reflecting  $f(t)$  on to the abscissa between the limits  $-\pi/2$  and  $3\pi/2$ .

The calculation yields the following FOURIER series for the demodulated characteristic:

$$(8) \quad f^*(t) = A_0 + \sum_{\nu} A_{\nu} \cos \nu \omega t + \sum_{\nu} B_{\nu} \sin \nu \omega t,$$

where

$$(9) \quad \left\{ \begin{array}{l} A_0 = (2/\pi) I_B, \quad A_1 = (8/3\pi) s a \cos \varphi_0, \\ A_2 = (4/3\pi) I_B, \quad A_3 = (8/15\pi) s a \cos \varphi_0, \dots, \\ B_1 = - (4/3\pi) s a \sin \varphi_0, \quad B_2 = 0, \\ B_3 = - (4/5\pi) s a \sin \varphi_0, \dots \end{array} \right.$$

### 1.4. Discussion of the Function $f^*(t)$

For any chosen deflection from the guide beam, the fundamental frequency and the harmonics can be derived from equation (8) in terms of amplitude and phase. The voltage and phase of the double fundamental frequency, which is given by the term  $A_2 \cos 2\omega t$  are independent of the position in space of the

flying object, and this phase may therefore be used, after frequency division, as a reference phase.

The amplitude and phase of the fundamental frequency are determined by the magnitude of the angular deflection  $\varphi_0$ . In particular, for the horizontal deviation transverse to the direction of the receiving antenna, i.e., for  $\varphi_0 = 0$  or  $\varphi_0 = 180^\circ$ ,

$$\text{fundamental} = \pm (8/3 \pi) s a \cos \omega t$$

and for vertical deflections in the direction of the receiving antenna, when  $\varphi_0 = 90^\circ$  or  $\varphi_0 = 270^\circ$

$$\text{fundamental} = \mp (4/3 \pi) s a \sin \omega t.$$

Therefore the amplitude obtained for the horizontal deflection is twice that for the vertical deflection, and a correction for this must be made in the evaluation.

Fig. 4 shows the amplitudes and phase relationships for the fundamental frequency and the first harmonic for a deflection  $\varphi_0 = 45^\circ$ .

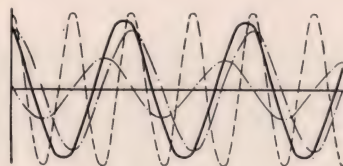


Fig. 4. Amplitude and phase of fundamental frequency and reference frequency at  $\varphi_0 = 45^\circ$

$$\begin{aligned} & \text{—} \quad \frac{8 s a}{3 \pi} \cos \varphi_0 \cos \omega t - \frac{4 s a}{3 \pi} \sin \varphi_0 \sin \omega t \\ & \text{---} \quad \frac{4}{3 \pi} I_B \cos 2 \omega t \end{aligned}$$

The amplitude of the fundamental frequency, which determines the magnitude of the error voltage, and therefore the sensitivity of the restoring signal for a deflection, is directly proportional to the magnitude of the deflection and to the sectional slope of the directive diagram.

However, the precision of the steering mechanism cannot be increased indefinitely by increasing the slope because, apart from the limits imposed by the physical dimensions of the antenna itself, the increase in slope is accompanied by an increase in diagram concentration. With too narrow a directive characteristic there is always a danger of the flying object moving out of the area of the main diagram and becoming located between the latter and the first side lobe. Signals that are out of phase by  $180^\circ$  are then given, which prevents the aggregate from being brought back into the guide beam.

As is described in more detail in section 1.2, the derivation of function  $f^*(t)$  was carried out under the very simplified assumptions that the directive diagram is cylindrically symmetrical with respect to its centre line and is also linearly inclined in the vicinity of the guide beam.

Herr ROCKSTUHL calculated the demodulated characteristic on the assumption, which approximates more closely to reality, that the diagram had the form of a BESSEL function of the first order. It was shown that the most favourable inclination of the diagram in the guide beam is obtained when an eccentricity of  $3^0$  is taken as basis. This was assumed in Fig. 1, and was actually the case for the WÜRZBURG-RIESE antenna.

In addition, the calculation shows that independence of the phase of the first harmonic is only true for small deflections from the guide beam. With greater deflections a small rotation of the reference phase also occurs. In this case the flying object would not return to the guide beam in a plane, but on a path curved in space.

In an application of the guide beam method to control in one guide plane only, e.g. guiding the V-2 in a horizontal plane, the reflector of the guide-beam transmitter had to be automatically re-adjusted in a vertical direction, otherwise a small transverse signal would have been given at large vertical deflections.

## 2. THE HV METHOD

### 2.1. Problems of Microwave Developments at the Beginning of the War

The priority which the German Air Force had for development at the beginning of the war required us to think firstly of the application of the method for the guiding of glide bombs.

For reasons of space the scanner diameter had to be less than 60 or 70 cm and therefore in order to have a beam width of at least  $10^0$  the transmitter frequency was fixed on the S band. But it is known that at that time there were no instruments and microwave tubes for developing the guiding system in this band.

Under acceptance of additional sources of error, it was possible to take over some measuring devices from the 50 cm band, for which, because of the radar project "WÜRZBURG" and the radio relays "MICHAEL", excellent measuring instruments were available at that time. But the greater part of the equipment had to be constructed by our own development group.

In the range of valves, neither a transmitting valve, nor an oscillator valve, nor a mixer valve nor a mixing crystal was available.

Therefore, in co-operation with Dr. FRITZ, who was in charge of the magnetron development at TELEFUNKEN at that time, a magnetron oscillator was developed, which gave a continuous-wave output of 3 to 5 watts. In the case of the mixing oscillator two possibilities were considered: either the mixing of the harmonics using the triode LD 1 as oscillator valve, or the direct mixing with the magnetron RD 2 Md, which at that time was in process of development. There were two alternatives for the mixer valve itself: the acorn diode SA 100 or the improved SA 102 and a mixing crystal, which had been removed from a longwave detector and built into a cavity resonator specially for the 10 cm band. For the remaining circuit technique the universal valve RV 12 P 2000 was available.

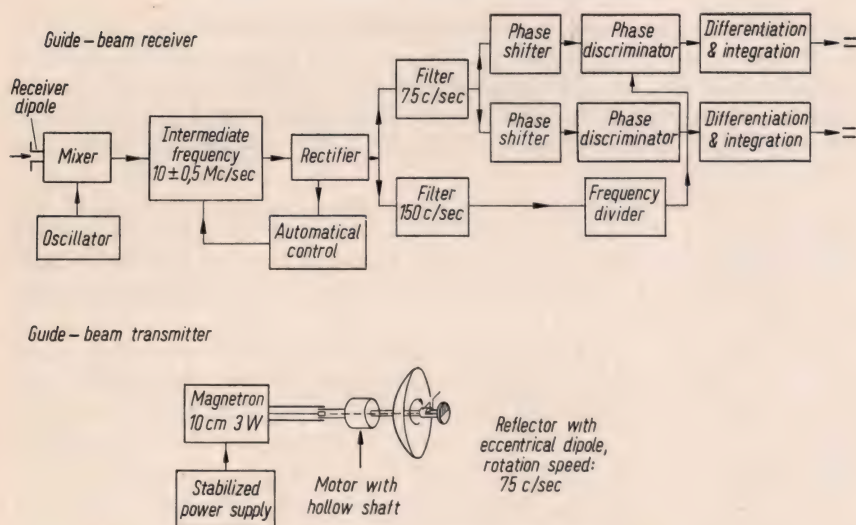


Fig. 5. The HV guide-beam system

With these available requisites and much skill in improvisation, the guide-beam apparatus HV, as it is shown schematically in Fig. 5, was developed.

## 2.2. The Guide-Beam Transmitter

The magnetron oscillator was used in continuous-wave operation and was connected by means of a capacitive coupling to the co-axial feed line formed by the hollow axis of the motor. The eccentric dipole was a smaller model of the primary feeder of the 50 cm radar equipment and was driven by a synchronous motor with a frequency of revolution of 75 c/sec.

## 2.3. H. F. and I. F. Parts of the Receiver

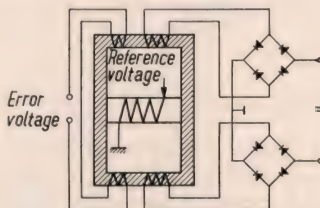
The absence of any kind of special valves for the high frequency and intermediate frequency circuits in the receiver resulted in considerable noise factors of the magnitude of 30 to 40 db. Overmodulation of the receiver when close to the transmitter would deform the modulation characteristic and produce false signals. Therefore the I. F. stage was provided with automatic gain control.

## 2.4. The Low Frequency Evaluation

After rectification, the reference frequency of 150 c/sec was separated from the fundamental frequency of 75 c/sec by means of filters. However, as phase comparison can be carried out only with voltages having the same frequency, the 150 c/sec frequency had to be halved in a frequency divider. This was carried out with the aid of a coherent oscillator. As is well-known, however, the phase of the output voltage of such a divider is uncertain by  $180^\circ$ , and therefore the oscillator had to be correctly set in phase before each control operation. The phase-sensitive rectifiers, or discriminators, which supplied the voltages proportional to the horizontal and vertical deflections, were placed after the fundamental frequency filter.

With two phase rectifiers, there are two possible methods of separating the fundamental frequency error voltage into horizontal and vertical components. Either the reference voltages can be supplied, through a phase changer, to the discriminators with a phase difference of  $90^\circ$ , or the error voltage itself can be transformed, through two phase changers in the horizontal and vertical channels, into two voltages shifted relative to each other by  $90^\circ$ , and compared to the same reference phase in the discriminators. In general, the latter procedure was chosen in the HV method, as the two phase changers also permitted the compensation of any unwanted phase shifts in the various amplifier channels.

A range of phase-rectifier designs was available. As their accuracy of phase governed by the accuracy of the given signals, they had to be designed with special care. An example of a rectifier, as used at the beginning, is shown in Fig. 6. The addition of the error voltage to the reference voltage took place through a three-phase transformer; the error voltage, unlike the reference voltage, being supplied in push-pull.



*Fig. 6. Phase discriminator with three core transformer*

The rectification itself took place in two bridge circuits, made up of cuprous oxide rectifiers. As stray capacities to earth can give rise to additional phase errors, special attention had to be paid to the symmetry of the winding.

The d.c. voltages, which were supplied by the two discriminators, and whose sign and magnitude provide a measure of the deflection from the guide-beam, were then specially treated in suitably dimensioned quadrupoles in order to prevent the building-up of control oscillations. Here they are differentiated with the aid of RC components and integrated with time constants appropriate to the mechanical and aerodynamical properties of the flying bodies.

## 2.5. HV Control Planned for Air-to-Ground Flying Objects

The control of a gliding bomb by means of a guide beam from an aircraft requires a stabilised suspension of the guide-beam antenna. Two methods of release were considered:

1. The antenna was situated in the nose of the aircraft; the start took place on approaching the target; the bomb-aimer held the guide beam pointed steadily at the target, whilst the aircraft itself endeavoured to keep as closely as possible on course, in spite of opposing defensive action.
2. The antenna was situated in the bath-tub gunners pit; the start also took place during the approach. After the release the aircraft turned around and whilst it was receding the bomb-aimer took up the flying object in the guide beam and directed it against the target.

Which of the two possibilities was finally decided upon is unknown, since no tests were made. At that time also, the notorious order, which stopped all further development in the cm wave region, and called for the conversion of this method to the 50 cm band was given. Because of this, the operation of air-to-ground control was out of the question, but it again became of interest when, at about the beginning of 1944, for equally well-known reasons, cm wave research was taken up afresh.

### 3. THE HAWAII II METHOD

The good experimental results obtained with the HV method in the laboratory suggested, at an early stage, that it could be used for controlling the ground-to-ground V-2 rocket. This operation was greatly simplified by the fact that the ground station could be taken over directly from the radar project. Before describing the instrument side in more detail, a brief description will be given of three methods, projected by the firm of TELEFUNKEN, for controlling the V-2.

#### 3.1. The Guiding Methods HAWAII I, HAWAII II, and ZIRKEL

Whilst the change-over from the HV method to the HAWAII II method was taking place, a guide-plane method HAWAII I, which operated in the VHF region, was being tested. It consisted of two dipole antennae erected at a certain distance from each other, their phases being reversed at a frequency of 50 c/sec. The radiation characteristic was strongly lobed, but it possessed considerable slopes, and the control gear was highly sensitive. However, as the diagram also depended upon the ground conditions, it had to be plotted anew for each firing site. This method, therefore, could only be considered for permanent sites.

The HAWAII II method was the guide-beam method with horizontal and vertical control which had been taken over from the HV technique and it operated on a 53 cm carrier wave. It was not brought into action before the end of the war, although measurements with a simplified modification for trial purposes were made with airborne V-2.

The ZIRKEL method was a HV method, operating in the 20 cm band, which also measured distance and speed fully automatically so as to give the combustion cut-off signal on reaching the desired values. As far as is known to me, a long series of individual instruments were fully developed for this purpose, but the complete system was never tried out.

#### 3.2. The Ground Equipment of HAWAII II

For the control of a surface-to-surface rocket, four quantities must be accurately known, viz. the horizontal and vertical signals and the distance and speed at the cut-out point.

The horizontal and vertical signals can be derived from the HV method. The distance was obtained quite easily by the use of an impulse-excited radar instrument of the "WÜRZBURG" series as a guide-beam transmitter. For this method it is, of course, completely immaterial whether this is in continuous wave

or in impulse operation, since only the response of the receiver to the envelope is evaluated. The technique employed with these radar instruments enabled distance measurements of extraordinary precision to be made.

The measurement of the speed was carried out from ground according to the DOPPLER principle. This however, required the additional installation of an airborne transmitter.

The measurements of distance and speed do not concern the actual guide-beam method and will not be described further.

The ground station consisted of a guide-beam antenna of 7 m diameter, the diagram having a width at half-maximum value of  $5^\circ$ , and an eccentricity of  $3^\circ$ . The output of the impulse-keyed transmitter was more than adequate to bridge

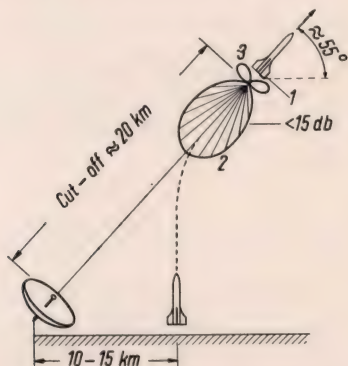


Fig. 7. Directional radiation pattern of receiving antenna

1 = Antenna dipole, 2 = Main pattern of receiving dipole, 3 = Side lobe

the approximately 20 km long path of the guide beam, in fact, it even led to difficulties in control at the receiver. The rocket was supposed to start vertically at about 10 to 15 km distance from the guide-beam transmitter, then be steadily brought into the guide beam by a pre-set steering device. During this, the pre-set steering was continuously weakened, so that a gradual change from pre-set steering to guide-beam control took place (Fig. 7).

### 3.3. The Aircraft Equipment

The receiving antenna had to be attached to one of the four control surfaces at as great a distance as possible from the beam, in such a manner that the aerodynamics of the missile would not be influenced. Moreover, its radiation characteristic had to be as rotationally symmetrical as possible in the rear direction, and screened as far as possible in the direction of the enemy (Fig. 7).

On the whole, the dipole antenna finally fulfilled these conditions; its side lobes were less than 18 db and the forward radiation amounted to less than 15 db.

The receiver was essentially similar to that shown in Fig. 5 for the HV method. In the 50 cm band, naturally, the valve situation was considerably more favourable, so that receiver sensitivities of about 15 to 17 db could be

obtained. An automatic frequency regulation was also under development; whether it was ever taken into use is beyond my knowledge. The I. F. amplifier had a band of  $25 \pm 0.25$  Mc/sec. In the low-frequency evaluation the phase rectifiers too were considerably simplified, and their accuracy improved. The

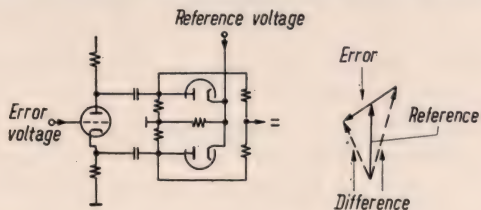


Fig. 8. Receiver dipole at stabilizer

places of the cuprous oxide rectifiers had been taken throughout by valves operating in push-pull. Fig. 8 shows such a rectifier stage; with the help of a phase-reversal stage the error voltage, in push-pull, is added vectorially to the reference voltage.

#### 4. THE CONTROL OF THE V-2

The control of the V-2, in both horizontal and vertical directions, was never carried out using the HAWAII II system, but for several trial launchings horizontal control was added to the pre-set steering as a fine correction; the vertical steering was always pre-set.

As was described in section 1.4, the reflector must be re-adjusted by hand, or automatically, in the vertical direction, in order to prevent wrong horizontal signals at large vertical deflections. This re-adjustment of the antenna could be carried out by impulse-reflection, as well as by a reverse operation of the HV method (transmitting dipole fixed, rotation of the receiving dipole) with the help of a transmitter on board the missile.

The principle of this fine-correction in the horizontal plane is shown in Fig. 9.

After differentiation and integration, the error voltage was not fed directly onto the rudder, but was transformed in a bridge modulator, with the aid of the 500 c/sec voltage of the power supply on board the missile, into a voltage

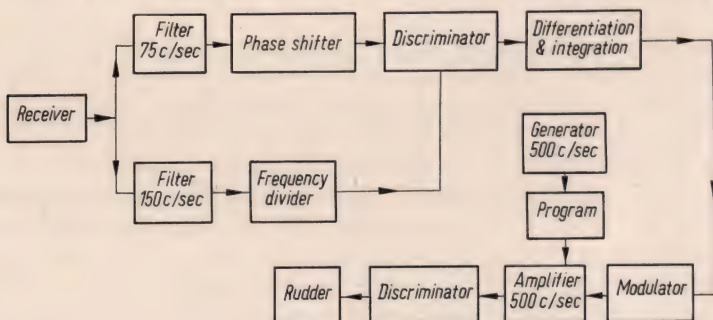


Fig. 9. Rudder system of V-2

having a frequency of 500 c/sec. This voltage was then mixed with the pre-setting component in an amplifier and, with the aid of a discriminator, was again changed back into a d.c. voltage. In this way the use of unstable d.c. amplifiers was avoided.

## 5. THE PRECISION OF THE GUIDE-BEAM METHOD

An accuracy of  $\pm 0.005^\circ$  was aimed at in the control of the V-2. This required extraordinary care in the electrical construction of filters, phase changers, phase rectifiers and modulators on board the missile, as well as stabilised voltages and the use of temperature-compensated elements. Very great demands were made on the accuracy of construction and the stability of mounting, of the reflector of the guide-beam transmitter. The ground reflection of the side lobes, as well as the changes of the refractive index by the exhaust gases of the rocket, also contributed to the deformation of the guide beam. The sounds and mechanical vibrations lead to valve microphonics and interfering modulations. The sources of error were, in part, of a physical nature (e.g., the curvature of the guide beam by atmospheric conditions) and could not be removed by technical means.

On consideration of all these factors it is questionable whether the state of technology at that time would have enabled such accuracy to be achieved.

A trial for testing the accuracy of the guide-beam method was carried out by TELEFUNKEN in 1942.

The experimental device was as follows: A 7 m reflector was firmly erected at an inclination of about  $10^\circ$ , on an open site near Groß-Ziethen. The course of the guide beam was flown by a Ju 52 carrying a complete guide-beam receiver, the output of which operated a crossed-pointer indicator. The indication was checked by a camera. A second camera with cross-wires was mounted so that it could take ground photographs in a vertical direction, and a third camera continuously checked the horizontal indication of the aircraft. The three cameras were released simultaneously and the guide beam, as measured, was traced onto a chart.

The tests showed that up to a distance of about 10 km, but especially in the vicinity of the transmitter, considerable disturbance of the guide beam was caused by ground reflections. In action, naturally, this disturbance is considerably less, as the reflector then has an inclination of about  $55^\circ$  in its final position, and the ground is irradiated by far less energy.

## 6. GUIDE-BEAM TEST INSTRUMENTS

The testing of a guide-beam method on site is time-consuming and costly. Therefore, for laboratory tests an instrument was developed which would best be designated as a guide-beam test oscillator, and which reproduced, to a very good approximation, the modulation characteristics as they occurred inside and outside the guide beam. Fig. 10 illustrates the principle of such an instrument.

A cylindrical or rectangular cavity *A*, which is coupled to a shielded H.F. oscillator in such a manner that it can revolve, has its fundamental frequency ( $TM_{010}$ ) excited by the latter. A coupling loop *B* revolves in the concentric magnetic field of the cavity with the frequency of revolution of the guide-beam

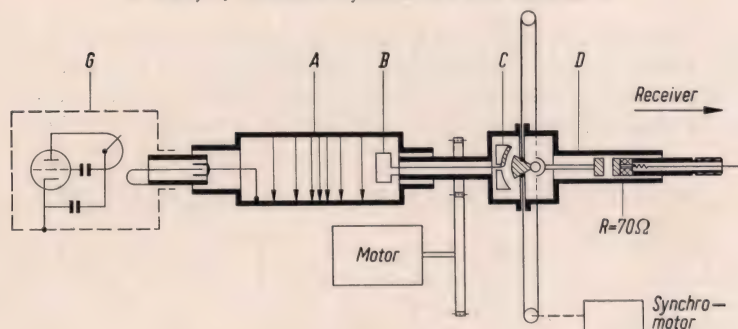


Fig. 10. Guide-beam test transmitter

A = Cylindrical or rectangular cavity

B = Coupling loop

C = Condenser

D = Damping device

G = High-frequency generator

diagram. Connected rigidly to this loop is a spherical condenser C, its metallic coating within the hollow sphere being designed so that for a certain angular deflection of the spherical sector, the same voltage differences occur as would be produced in space by an overlapping of the directive diagrams corresponding to a deflection from the guide beam. Fig. 11 shows the rotating and the fixed condenser coatings in the state corresponding to the position in the guide beam, i. e., the error voltage is zero. The spherical sector is connected, through a spherical joint to a damping device D; this permits the reproduction of the weakening of the energy received during the flight. The resistance R replaces the source resistance of the receiving dipole.

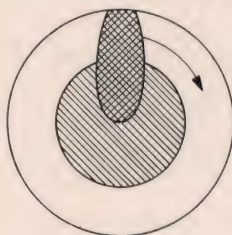


Fig. 11. Ball shape condense with copy of antenna pattern

With the coupling loop in a position relative to the field as shown, a deflection of the spherical sector in the plane of the illustration would correspond to a horizontal deflection of the missile; a deflection perpendicular to the plane of the illustration, to a vertical deflection.

However, any chosen angular deflection may also be brought about by a simple displacement of the spherical sector within the plane of the illustration if, at the same time, the cavity A is rotated through the corresponding angle.

With this instrument, therefore, every deflection from the guide beam can be represented in terms of amplitude and phase.

In tests carried out on oscillating tables, for the determination of the differentiation and integration constants, it was also possible to operate the spherical sector by means of a SELSYN-emitter-receiver-system, so that the signal-current proportional to every deflection was automatically adjusted to the right value.

# GUIDANCE OF SURFACE-TO-AIR MISSILES BY MEANS OF RADAR

KARL H. SCHIRRMACHER \*

## 1. STATEMENT OF THE PROBLEM

The exigencies of war and experience acquired previously in related fields of work encouraged the authorities in the development of anti-aircraft missiles (hereafter called a. a. missiles).

This decision meant at that time a further step forward in the techniques of guided missiles.

Without implying that the techniques employed in the V-2 and gliding bomb were less complicated, the fact that the target itself had a greater velocity was an additional factor to be taken into account, and the peculiarities of the a. a. missiles and their guidance systems are probably due to this fact.

Guided missiles have to be guided and controlled from the ground towards flying targets and exploded in their vicinity. The combat space around the launcher is bell-shaped with a diameter of 60 km (approximately 40 miles) and a maximum height of 15 km (10 miles). According to estimates made at that time, warheads were lethal against big bombers at a distance of approximately 50 m (150 ft). Taking the greatest range of 30 km (20 miles), 50 m (150 ft) corresponds to  $\frac{2}{16^0}$  or 2 mil, where 1 mil is  $\frac{1}{1000}$  of a radian. This is the accuracy requirement, which will be dealt with later. Furthermore, it was required to launch one missile every minute from a battery, which consisted of 4 launchers. In the case of the coordinated fire of several batteries, it was required to control 20 missiles simultaneously. This fixed the necessary frequency bands used. Attack had to be possible when the target was not visible, a requirement which then was dictated by weather considerations rather than by high-altitude flying.

It was obvious, that high-frequency techniques provided the only solution to the problem. As a result, in this area of the guided missiles work cooperation between the electronics engineer, the aerodynamicist, and the control-mathematician was most pronounced.

Apart from the proximity fuse and the warhead, the guiding equipment was composed of three main parts:

1. the target location,
2. the missile location, and
3. the guiding control proper.

This paper deals mainly with items 2 and 3 and, where necessary, the links between 1 and 2.

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## 2. EXISTING TECHNIQUES

### 2.1. Command-Link Systems

As shown already in a previous paper, considerable experience, technical as well as military, had been gained from the work on gliding bombs and it seemed natural to take these techniques and modify them for the task. The "KEHL-STRESSBURG" system was in mass-production. It had been developed for the Hs 293 and the use of a mark-space ratio modulation had not only the advantage of a high control accuracy, but was the most suitable for the spoiler control of the Hs 293. The modifications which proved necessary and the difficulties encountered with this new development will be discussed later.

### 2.2. Guiding-Beam System

A variant of the H and V system which had been developed for the missile A-4 (V-2) appeared to be the most suitable solution for the problem of the location of the missile. The H and V method is a guiding beam system which provides for a transmitter on the guiding and a receiver on the guided part, or vice versa. The information, i.e. the deviation from the beam, is obtained by modulating the radiation with a rotating aerial, preferably on the ground, in such a way that an alternating current of the frequency of the rotation is generated when the missile deviates from the beam. The amplitude of this alternating current is a measure of the amount of the deviation and its phase, relative to that of a reference signal, represents the direction (perpendicular to the beam) in which the displacement occurs. The H and V method employed linearly polarized aeriels and this fact presented some difficulties, as will be shown later.

## 3. THE PLAN FOR DEVELOPMENT

### 3.1. The Four Different Missiles

Difficulties arose in applying the available techniques because of the fact that four missiles were under development. These differed in many respects. The missiles were: "WASSERFALL", "RHEINTOCHTER", "SCHMETTERLING", and "ENZIAN". The first two were supersonic missiles with a cruciform shape; the latter two were subsonic and had aircraft-like bodies. The propulsion was different, too.

Table 1 shows the situation in detail. The different control methods should be especially noted. These differences determined primarily the modifications to the command link. Furthermore, they affected the design of the aeriels, which had to be fitted to the aerodynamic form of the missiles.

### 3.2. General Trend of the Development

It was the intention to carry out the design in such a way that any new development and improvement could be made to the existing equipment without developing the complete equipment again. The general plan was as follows.

Initially, location of the target as well as the missile should be achieved by optical means and the command link by means of the "KEHL-STRESSBURG" system. Later the command link was to be replaced by the "KRAN-BRIGG" system,

which employed a higher carrier and higher modulation frequencies. The next step would have been the changeover to radar systems for the target and missile location. Initially a wavelength of 50 cm was projected for the radar systems, but the use of 10 cm wavelength was envisaged.

In the first instance the object could have been achieved with this programme. In the course of the development, however, it became apparent that it would be desirable to use homing heads for the last part of the trajectory. In order to guarantee a smooth transition to such techniques, it was planned initially to retain the human operator. However, for the last part of the trajectory a television picture, which displayed the target as seen from the missile was to be used instead of the optical or electronic display. The experience gained with these techniques would have contributed to the development of homing devices.

#### 4. THE PROPOSED SYSTEMS AND THEIR OPERATION

The "Ziel-Deckungs-Verfahren" ("three-points method") was made the underlying principle of the guidance. A target is located by the radar; the coordinates of the target relative to the launching point are therefore known continuously. The missile is launched in the direction of the target and appears on the display of the missile location equipment. The missile is now steered on to the line joining the observation point and the target and kept there by the human operator by means of the radio command. The rocket must now at some time pass near to the target and the proximity fuse would be triggered.

In addition to the apparatus mentioned, computers, which connected the various phases of the trajectory with the shortest possible transient times, were also part of the equipment.

##### 4.1. Launch from a Projector

As the first example the equipment for projector-launched missiles may be described. There is no point in describing separately the optical equipment, for it follows the same principle.

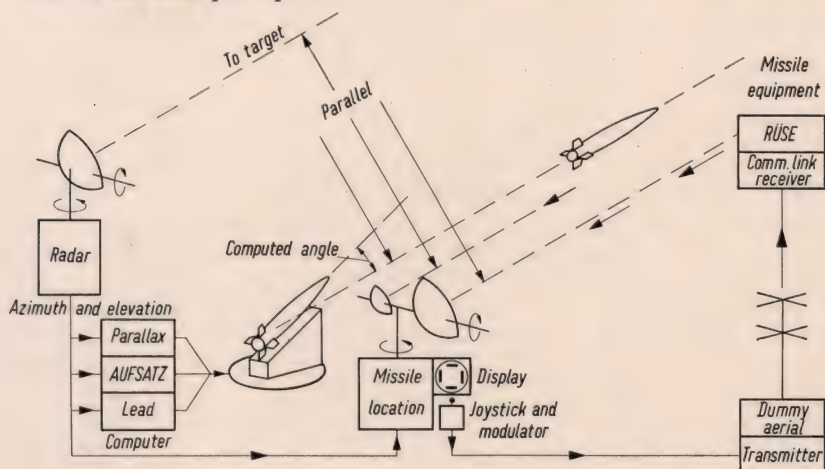


Fig. 1. Guidance system for projector-launched missiles

A radar of the "WÜRZBURG" or "MANNHEIM" type with a reflector of the "RIESE" type served as target location equipment (see Fig. 1). The angles were transmitted to the reflector mounting of the missile location gear, which had a "MANNHEIM" type aerial and reflector. At the same time the angular information was transmitted to the computers, in this case the "lead", "drop" and "parallax" computer, which remotely controlled the projector after the computation was completed, and the  $\tau$ -computer, which turned the base of the joy-stick. Radar, missile location and projector were kept nearly parallel. The firing itself was carried out by the joy-stick operator, who received a verbal command from the Officer-in-Charge of the battery. After the launch the single-stage oscillator "Rüse", which was fixed to the missile, was first picked up with the small reflector. After some seconds, the reception was switched to the main reflector. The direction of the radius vector was displayed to the

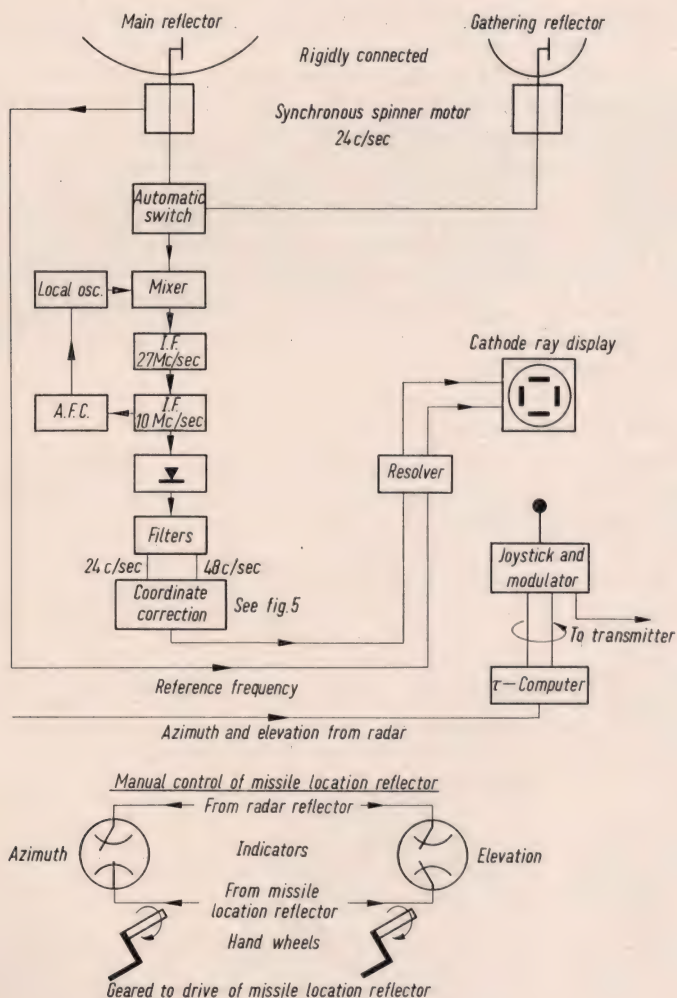


Fig. 2. Block diagram of missile location equipment

operator on a cathode-ray tube, and the operator's position was arranged in such a way that he was able to use a telescope instead, visibility permitting. The radar, too, could be directed by optical means.

The optical system had the distinction that target and missile location were fixed on the same mounting, in order to exclude transmission errors.

The aerial system of the command-link and its transmitter were placed apart from the rest of the equipment and were remotely controlled. Before the launching the transmitter was connected to a dummy aerial in order to avoid unnecessary radiation, the changeover to the radiating aerial being made at the moment of launching. The aerial was circularly polarized in order to maintain transmission during roll movements of the missile, as the latter was equipped with a linearly polarized aerial. For details of the missile location see Fig. 2.

## 4.2. Vertical Launch

The vertically launched "WASSERFALL" required a somewhat different arrangement. The radar operated in the same manner as in the case of the projector-launched missiles (see Figs. 2 and 3), but the angular information was transmitted to the "Einlenk" computer. The reflector for missile location was

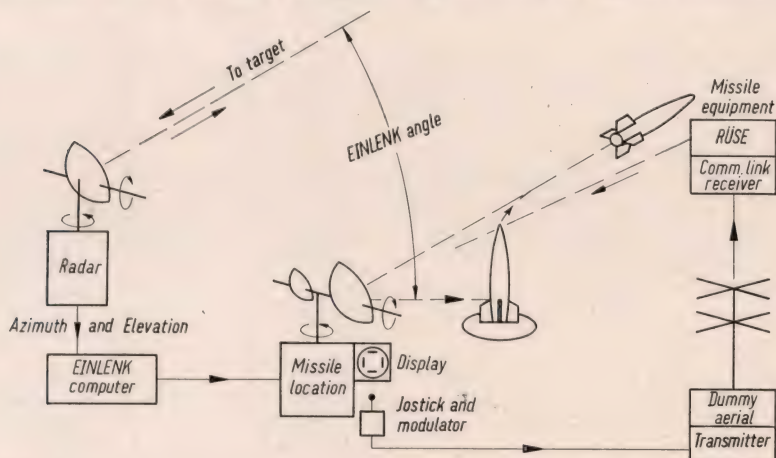


Fig. 3. Guidance system for vertically launched missiles

pointed towards the launcher up to the moment of launching. At that moment the "Einlenk" computer started to control the aerial for the missile location. In the shortest possible time the missile location reflector had to be directed into a position parallel to the radar. This was achieved by means of the "Einlenk" computer in such a manner that the movement of the missile location reflector depended not only on the movement of the target but also on the (previously known) manoeuvrability of the missile, in order to avoid excessive demands on the missile.

A parallax computer, and a  $\tau$ -computer were necessary as well as a device for turning the base of the joy-stick before the launch in order to maintain the right sense of the joy-stick movements. The "WASSERFALL" itself was not turned before launching.

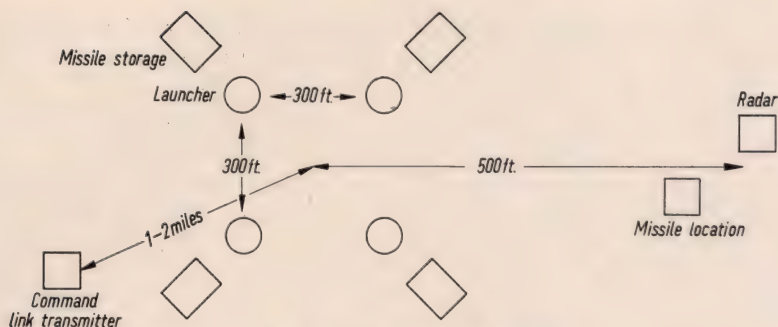


Fig. 4. Plan view of a battery

One battery consisted of four launching platforms or projectors (see Fig. 4). The distance between them was approximately 100 m (300 ft) and the guidance equipment was placed approximately 150 m (500 ft) from the centre of the square of the launchers.

#### 4.3. The Guidance Systems under Development

Table 2 shows the guidance systems under development and their code names and application. The table needs no further comment.

Table 1. Some data on the missiles.

Type	Launch	Propulsion	Control	Average velocity	Computers
WASSER-FALL	vertical	single stage liquid fuel	cartesian; proportional rudder	$\approx 500$ m/sec $\approx 1600$ ft/sec	$\tau$ -computer Einlenk-computer
RHEIN-TOCHTER	projector	two stage solid fuel	cartesian; proportional rudder at the front of the missile	$\approx 450$ m/sec $\approx 1500$ ft/sec	$\tau$ -computer parallax-computer lead-computer "drop"-computer
ENZIAN	projector	solid fuel boost; liquid fuel sustainer motor	polar; proportional rudder	$\approx 200$ m/sec $\approx 700$ ft/sec	$\tau$ -computer parallax-computer lead-computer "drop"-computer
SCHMETTERLING	projector	solid fuel boost; liquid fuel sustainer motor	polar; spoilers	$\approx 250$ m/sec $\approx 850$ ft/sec	$\tau$ -computer parallax-computer lead-computer "drop"-computer

Table 2. The guidance systems in development.

Code name	Target location	Missile location	Command link	Remarks
BURGUND	optical	optical	KEHL-STRASS-BURG (COLMAR) 60 Mc/sec	3 different versions 1. WASSERFALL 2. RHEINTOCHTER 3. SCHMETTERLING and ENZIAN
ELSASS	radar Mc/sec 600 Mc/sec	radar (RÜSE) 600 Mc/sec	KEHL-STRASS-BURG (COLMAR) 60 Mc/sec	3 different versions
FRANKEN	optical	optical	KOGGE-BRIGG 1250 Mc/sec	3 different versions
BRABANT	radar 600 Mc/sec or 3000 Mc/sec	radar (RÜSE) 600 Mc/sec	KOGGE-BRIGG 1250 Mc/sec	3 different versions
PARSIVAL	optical	optical	KEHL-COLMAR 60 Mc/sec	no computers; for SCHMETTERLING only
LOHEN-GRIN	optical	optical	KOGGE-BRIGG 1250 Mc/sec	no computers; for SCHMETTERLING only
ELSASS-LOTH-RINGEN	radar 600 Mc/sec	radar (RÜSE) 600 Mc/sec	KEHL-STRASS-BURG 60 Mc/sec	like ELSASS with the additional use of television equipment for the latter part of the trajectory

## 5. THE EQUIPMENT IN DETAIL

### 5.1. The Location of the Target

As already mentioned "MANNHEIM" and "WÜRZBURG" equipment was to be used as the radar. The normally used 3 m (10 ft) diameter reflector was replaced by a reflector of 7 m (23 ft) diameter and the mounting was altered accordingly. This aerial system was called the "RIESE" (giant).

The radar itself will not be discussed in detail; only a few relevant remarks may be made. The wavelength used was 50 cm. In the course of the development of the a. a. missiles intensive investigations were carried out by Major HOFFMANN of the FLAKVERSUCHSSTELLE REINICKENDORF to establish the accuracy of the radar. The result of these experiments with carefully selected crews was that a mean value of 1.5 mil could be guaranteed and sometimes a mean of 0.75 mil had been obtained. It should be noted in this context that German radars were not automatic; the values had to be adjusted manually.

## 5.2. Location of the Missile

For the missile location the H and V guidance principle was used. This method will be dealt with in another paper. The principle was mentioned earlier. It will be convenient, however, to enlarge the subject a little more in order to be able to explain the modifications made to the original concept.

A rotating aerial, which acts as the modulator, is placed in the transmission path. Since "MANNHEIM" aerial systems were used the scanning-frequency was 24 c/sec. If now a reference phase is established, for instance, in a certain position of the dipole, resolution of the received signal can be obtained in respect to amplitude and phase.

In the case of the a. a. missiles the transmitter was placed in the missile and the radiation was received through the scanning aerial on the ground. After demodulation one alternating current voltage was obtained of the scan frequency and another of twice that frequency, caused by the linear polarization of the transmitting aerial.

If now the system presented a true picture of the geometrical situation, the first-mentioned voltage converted into rectangular components could be used as quantity indicating the deviation off the beam.

As is shown in the paper by Dr. SCHÄFER, the displacement obtained in one direction is only half that obtained in the other direction, if the "RÜSE" transmitter in the missile and the ground equipment are fitted with linearly polarized aerals. In particular one obtains a correct indication from a displacement occurring perpendicular to the orientation of the airborne aerial, and half the correct indication from a displacement in the direction of the dipole axis.

If, therefore, a missile is moving around the beam with a fixed displacement, the locus of the received vector is not a circle, as expected, but an ellipse with the axis ratio of 2 to 1. The major axis is perpendicular to the dipole and its orientation depends therefore on the relative position of the roll-angle of the missile with respect to the coordinate system on the ground. With missiles of the A-4 (V-2) type the effect can simply be compensated by an amplification factor of 2 in the appropriate component, as in this case the missile-borne co-ordinate system does not change in respect to the ground. This is not so in the case of a. a. missiles.

A comparatively simple solution of the problem appears to be the use of circularly-polarized aerals. Difficulties, however, arise in the 50 cm band to devise simple aerial structures which fulfil at the same time the requirements of an accurate circular polarization and a relatively great bandwidth.

Therefore the following solution (illustrated by Fig. 5) was chosen. From the output of the receiver two vectors can be obtained, a vector indicating the position of the missile with respect to the beam and a voltage of twice the scanning frequency, which is always in the direction of the major axis of the ellipse. The true vector should describe a circle, but the measured vector describes an ellipse. The vector difference between these two is always parallel to the minor axis of the ellipse.

The measured vector in general is shifted by an angle  $+\psi$  towards the vector of twice the scan frequency or, in other words, towards the major axis of the ellipse. If now the measured voltage beats with the voltage of twice the

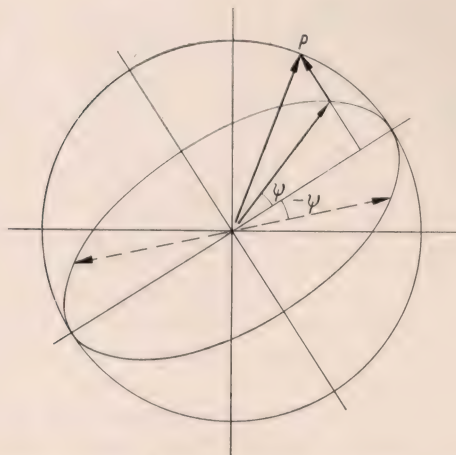


Fig. 5. Diagram to explain the modification of the H and V method

scanning frequency, a voltage of the scanning frequency with the phase angle  $-\psi$  and the same amplitude results. This vector, with changed sign, added to the originally-measured vector produces exactly the difference vector between required and measured vector, but with twice the amplitude. A further vector addition results in the required vector.

The resolution of this vector into two components is normally obtained by means of phase-bridges. In this case the resolution could be achieved comparatively easily as the necessary reference frequency was available on the rotating shaft. The measured voltage was fed into the grid of a valve, the output of which was connected to four condensers in turn. The charging of the condensers was done through the contacts of four relays, which were operated by four notches on the shaft of the motor driving the dipole. The voltage between two condensers served to deflect the cathode-ray of the display tube. The direct current voltage had to be slightly filtered. The layout of these filters had not been decided finally at the end of the war.

The normally used "MANNHEIM" reflector was 3 m (10 ft) wide, and had a linear angular range of  $\pm 6^\circ$ . The gathering reflector was much smaller and its angular range was  $\pm 20^\circ$ . The receiver was automatically tuned.

The scanning frequency of 24 c/sec was not particularly advantageous. Therefore improved scanning aerials of 75 c/sec were under development. This new equipment would have eased the difficulties arising in the design of 24 c/sec circuits.

### 5.3. Additional Developments for the Command-Link Systems

The "KEHL-STASSBURG" equipment has been described in another paper. In this paper only its modifications and improvements and the reasons for them shall be mentioned.

The control signal for one co-ordinate was obtained by switching two different modulation frequencies periodically. The difference of the time intervals of the modulation constituted the command.

The concept of the HENSCHEL control systems provided for the direct application of this alternating signal to the spoilers. In the early versions of the equipment the switching was done by rotating cylinders. "WASSERFALL" and "RHEINTOCHTER", however, required proportional rudder deflection. This necessitated additional rectification of the switching frequency and filtering.

In this case, therefore, the switching frequency had to be increased. A frequency of 25 c/sec instead of the original 5 c/sec was chosen. The original modulation frequencies were 1, 1.5, 8 and 12 kc/sec. These were increased to 6, 9, 13 and 16 kc/sec. These frequencies must not generate combination frequencies which will interfere with any of the four channels. This condition entails the disadvantage that the frequencies cannot be easily generated from a common source by mixing. They are therefore not held firmly in phase relative to each other, and this means that the individual time intervals are not clearly defined.

Mechanical generation of the switch frequencies was replaced by a relay circuit, the "KLAPPER", which was in development with STASSFURTER RUND-FUNK AG. For later stages an equivalent electronic circuit was under development by TELEFUNKEN.

The main problem presented by the development of this piece of equipment was the demand for an extremely high zero output accuracy. A triangular waveform was generated in the "KLAPPER", which being symmetrical in respect to zero had a positive and negative section, which operated the relays switching the modulation frequencies. The direct current voltage proportional to the displacement of the joy-stick, the "KNÜPPEL", shifted the symmetry of the triangular waveform, so generating the difference in the time intervals.

In order to avoid distortions, the modulation characteristic of the transmitter had to have a large linear portion. Since two of the modulation frequencies were present at any instant, the maximum possible modulation percentage was allocated in equal parts to the two frequencies. For the calibration a modulation percentage indicator was indispensable.

The power of the "KEHL" transmitter was 50 W. The 20 channels were displaced by 100 kc/sec with the average carrier frequency of 60 Mc/sec; the total bandwidth was 2 Mc/sec or 3%.

In order to replace this system at a later stage, the "KRAN-BRIGG" system using an average frequency of 1250 Mc/sec was under development. The separation of the channels was 6 Mc/sec, so that the aerials had to have a bandwidth of 10%.

Two different joy-sticks had to be designed, one for the polar co-ordinate system and one for the cartesian co-ordinate system. The first of these has been mentioned in a previous paper. The joy-stick for the cartesian system consisted of two potentiometers orientated perpendicular to each other.

Although it really belongs to the field of control techniques, it might be mentioned here that it was not quite settled at the end of the war whether the joy-stick deflection should simply move the rudder proportionally, which would have resulted in a lateral acceleration, or whether the joy-stick command should cause the missile to assume a new direction. The latter method would probably have necessitated a differentiation of the joy-stick command.

It may be assumed that, depending on the missile, a compromise solution would have been found.

#### 5.4. The Required Computers

The computers may be mentioned only briefly. These are listed in Table 2. The purpose of the "Einlenk" computers has been mentioned already. For projector launched missiles the "Aufsatz" computer and the "Vorhalt" computer were necessary. The first compensated for the loss in elevation angle after launch. The missile has not accelerated to its full speed shortly after launching and consequently falls. This error depends on the elevation angle.

The "Vorhalt" or lead computer was necessary because the missile could only be launched from a fixed projector. At the moment of launching at first the projector was set still and the launching itself took some time. The necessary lead angle was provided by the computer.

The parallax between target and missile location beams was also taken into account by a separate computer.

All guidance systems include a  $\tau$ -computer. This choice originated from the fact that the roll reference was space-fixed. Some time after launching the missile roll reference was, relative to the ground co-ordinate system, at an angle

$\tau = \int_0^t \dot{\sigma} \sin \gamma \, dx$ , where  $\dot{\sigma}$  is the angular velocity in azimuth of the reflector mounting and  $\gamma$  its angle of elevation. This computation was located in the missile location system. It was obtained by electro-mechanical means to allow the mechanical turning of the base of the joy-stick.

Further corrections of the demanded guidance signal consisted of a modification of the elevator demand of the missiles steered on a polar co-ordinate system, depending on the angle of elevation and the necessary co-ordinate transformation at the base of the joy-stick before the moment of launch of the "WASSERFALL".

#### 5.5. Ground Aerial Mountings

Alt-azimuth mountings have the disadvantage that around them a cylindrical space with a cardioid base exists, in which a target cannot be lock-followed. Apart from the co-ordinates of the target this effect depends on the maximum possible angular velocity in azimuth.

For the mountings to be used at later stages, for which the scanner of higher r.p.m. was to be incorporated, provision was made for three controllable axes. The firm of "KREISELGERÄTE" was in charge of the design. The idea was to tilt the vertical shaft; this tilting motion being activated when the azimuthal angular velocity approached the critical value.

With respect to the "MANNHEIM" reflector mounting as used for the missile location, one point must be mentioned. This equipment was planned for its original use as a remote-control transmitter. Its mechanical design did not permit it to be controlled remotely from the radar: manual control was necessary. The angular difference between the two positions was displayed and the operator had to keep the difference angle at zero.

#### 5.6. Airborne Equipment and Aerials

The specification for the airborne equipment stressed the necessity of small weight and a minimum of handling and pre-launching service. Much work was

devoted to the design of the airborne aerial systems. The aerial for the "STRASSBURG" receiver was a dipole which in most cases was fixed to the rear edge of the fins. The small transmitter "RÜSE" and its aerial were constructed as a single unit. The simplest solution for this design was found in the "ENZIAN" for which this equipment was located within the wing and therefore did not contribute to the drag. For the "WASSERFALL" the "RÜSE" was sandwiched between the skin of the fins, and for the "RHEINTOCHTER" it was housed in a gondola and fixed at the outer edge of fin.

Extensive and experimentally difficult measurements were conducted by Dr. ZUMBUSCH using models as well as the missile itself in order to establish the form of the aerial polar diagrams.

The "STRASSBURG" receiver was of the superhet type and contained approximately 30 valves. The "BRIGG" receiver was also a superhet but used for the intermediate frequency stages an amplifier working on the principle of super-regeneration. The version in production at the end of the war contained 5 valves. The airborne equipment for guidance would have contained 6 valves including those in the "RÜSE".

## 5.7. Advantages and Disadvantages of the Proposed Systems

The systems described have the great advantage of being flexible with respect to modification. For instance, so far the question of an automatic guiding system, beam riding, has not been touched. The layout of the equipment would have permitted the changeover to such a system without great modifications. The solution would have been to connect the display voltage of the missile location to the input of the command-link system through damping filters. The airborne equipment would have remained unchanged.

One disadvantage of the system might be mentioned here. The receiver is looking towards the missile and therefore towards the target. The equipment, therefore, is susceptible to external interference.

## 6. TRIALS

### 6.1. General Survey of the Experimental Work

It should be pointed out that experimental work strongly affected the speed of the development in a favourable manner.

Firing trials devised to obtain aerodynamic and control data were used to test the airborne guidance equipment and the operation of the complete system.

At first the guiding beam system was tested by using a "RÜSE" in the missile and by receiving it on the ground in the proper way. Most of these experiments were carried out with the "RHEINTOCHTER", on a smaller scale also with "WASSERFALL" and later with "ENZIAN". The "RHEINTOCHTER" trials were of particular interest, as this missile had the highest initial acceleration (approximately 20 to 40 g). All these trials yielded some information about the interference encountered.

Later control experiments were carried out with "WASSERFALL", "SCHMETTERLING", and "RHEINTOCHTER". These experiments were satisfactory.

Up to the end of the war no evidence had been obtained which would have required alteration of the original concept.

Shortly before the end of the war a few television trials were carried out with the "RÜSE" as transmitter. Excellent results were obtained up to ranges of 20 miles.

The guiding beam system was also tried out in connection with the Me-163 fighter. The receiver output of the "STRASSBURG" was displayed to the pilot on an instrument. The fighter was located on the ground by means of the "RÜSE".

## 6.2. Interference and Noise

The interferences encountered may be mentioned shortly. These are caused by various effects, absorption and reflection on the flame, static charges, vibration and microphony. To mention the results first, it appears that the interference did not seriously affect the functioning of the equipment.

Many attempts have been undertaken to measure the reflection and attenuation due to the jet flame. No conclusive result was obtained. Perhaps the most exact experiments were carried out by Dr. ZUMBUSCH on a WALTER motor. The results showed the effects of this kind of interference, but that they were smaller than those due to vibration and microphony. Lack of time did not permit further investigations.

With regard to the microphony, frequency spectra were taken. It was found that the excited frequencies were different from the frequencies used in the system. This admittedly somewhat crude result was then regarded as satisfactory. The airborne equipment was mounted shock and vibration proof and packed in glass-fibre.

An interesting result was obtained by the comparison between the received guiding beam information as received from the "WASSERFALL" on the one hand and the "RHEINTOCHTER" on the other. Interference in the "RHEINTOCHTER" trials were of comparatively small amplitude and did not seem to exceed 200 c/sec, but the amplitude was much bigger and frequencies of more than 1000 c/sec have been observed with the same type of trial with the "WASSERFALL." The result was then explained by the different structure of the two missiles. It was unavoidable using a single dipole that the whole structure or at least part of it did not contribute to the shape of the radiation pattern. The "RHEINTOCHTER" was practically built from one solid piece of metal, whilst the "WASSERFALL" was covered with riveted metal sheets in the aircraft design fashion. It was surmised that if the contact between the sheets was not perfect random changes of the contact contributed to the interferences.

Furthermore, it was observed in the "RHEINTOCHTER" trials that the interference practically vanished as soon as the sustainer motor went out.

## 6.3. Accuracies

The missile location itself did have an accuracy better than  $1/16^0$  or 1 mil, assuming interference of the order of 20% modulation depth. The error of the radar was in the order of 1 mil. The manual follow procedure of the missile location reflector contributed at least  $1/2$  to  $3/4$  mil. On the whole the requirements would have been fulfilled. One has to take into account, however, that a complete test of the equipment against flying targets had not been carried out.

## 7. CONCLUSIONS

Shortly before the war ended, the development, design and construction work of the optical version of equipment for "SCHMETTERLING" was completed and the equipment was shortly to be used. The radar equipment had been designed and was in the process of construction.

The whole development was only made possible by the close co-operation of all concerned in the project. Although the responsibility for the guidance systems rested with TELEFUNKEN, mention must be made of the following: the Research Establishment at Peenemünde, the FLAKVERSUCHSSTELLE, the Air Ministry, the organizations in charge of the missile development at Peenemünde, the firm of RHEINMETALL-BORSIG, the HENSCHEL Aircraft Company, and the MESSERSCHMITT Aircraft Company, all of whom greatly contributed to the development of the guidance system.

The equipment could not be subjected to the great test, namely, the use for which it was developed. But from a technical point of view the results, which were obtained in the space of two years, are noteworthy.

# THE PHYSICAL AND TECHNICAL DEVELOPMENT OF INFRARED HOMING DEVICES

EDGAR W. KUTZSCHER \*

## 1. INTRODUCTION

In the summer of 1944, a report on the state of the art of homing devices based on radar, acoustics or infrared radiation and their possible use in various missiles was prepared for the German Air Ministry by Mr. THIERY, Mr. KLEIN and others associated with the LUFTFAHRTGERÄTEWERK, Hakenfelde. In the summary of their report, the authors made the following statement:

"The use of radar homing devices would be the most universal approach for guiding a missile during the last phase of its flightpath. The development of the radar homing device, however, is far behind that of other methods and, in addition, such a device would be very complicated. Homing devices based on an acoustic method will be ready in the very near future. Their use, however, is limited, especially if the missile is propelled during its entire flight. The development of infrared homing devices is most advanced. Such a device is indeed the simplest approach for solving the problems."

The recommendations made in this report helped to further an extensive programme for the development of infrared homing devices intended for an early practical use. Actually, these devices have not been put into operational use. Various models, however, were ready in the laboratories and waiting for field tests at the end of the war. People concerned recognized that many system and installation problems were not solved at that time and that much work had still to be done.

The development of these infrared homing devices was based on comprehensive efforts to use infrared methods and devices for various military applications. Basic laboratory investigations necessary for feasibility studies and for the technical development of military systems started early in the 1930's. This work on basic problems was concerned with:

calculations and measurements on the emission of special infrared transmitters and with the total energy and spectral content of the infrared radiation emitted by various targets of military interest;

studies of the attenuation of infrared radiation by the atmosphere;

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optical problems;

the design of special optical systems, including the development of new optical materials suitable for lenses and windows;

the creation of new infrared detectors;

investigations of the influence of background radiation;

theoretical and experimental work on methods for scanning optical fields and for modulating infrared radiation.

The research carried out in this field resulted in the successful design and use of practical infrared devices for various military purposes. It was possible to detect and track a target, whenever its temperature was sufficiently different from that of its background. For example, a ship, the average temperature of which may be only a few degrees higher than the ambient temperature, and the maximum temperature of which may be approximately from 50° to 70° C higher than the ambient temperature, could be detected and tracked up to distances of 10 to 35 km depending on its size and the weather conditions. During the war, devices of this kind were in practical use on the French coast. As the difference between the temperature of the target and the ambient temperature increases, detection and tracking become easier, and the tracking range increases accordingly.

## 2. THE PASSIVE AND ACTIVE INFRARED METHODS

The method of detecting a target by means of the infrared radiation emitted by the target itself is called the passive infrared detection method. Targets may also be detected by means of the so-called active method which is similar to radar in that an infrared beam is sent out by using a special transmitter and the energy reflected by the target is received.

The utilization of the infrared radiation emitted by the target itself has a great advantage because it permits the design of a very simple detecting device and does not require a transmitter.

The operation of the passive infrared device is such that the enemy is not aware of its operation. This is, of course, of great technical and military significance, in that detection and tracking do not lead to evasive action by the target. Further, it is very difficult to jam the infrared detectors specifically since the time interval in which the homing device is working is not known. Camouflaging the target by, for instance, painting it with materials having a very small reflecting power in the infrared is useless because reflection plays no part in the passive method of tracking or detecting. A decrease in the amount of infrared radiation emitted by the target aids in preventing its detection by an infrared device, but experiments have established that such countermeasures are difficult, particularly if the efficiency of the power plant is to remain unimpaired.

Furthermore, in addition to the tactical advantage of the passive method, the efficiency of the passive system is far superior to that of the active system. For example, the infrared radiation emitted by a three-engined bomber in the wavelength region of approximately 0.8 to 3.5  $\mu$  was found to be from 1000 to 10,000 times greater than that reflected in the active method when a searchlight

with a special infrared filter and a mirror of 150 cm diameter was used. This comparison is based on a distance of approximately 10 km. The advantage of the passive over the active method grows with increasing range.

### 3. INFRARED RADIATION EMITTED BY MILITARY TARGETS

For the development of a passive infrared device, especially in considering the choice of a detector element and means to avoid interfering background radiation, accurate data on the infrared radiation of aeroplanes, missiles, ships, ground installations, or any other possible targets are necessary. It was therefore decided to make measurements of the radiation and its spectral distribution at various azimuth and elevation angles around the targets in order to determine their vertical and horizontal radiation patterns.

As a first approximation a target may be considered as a black or grey body radiator. Fig. 1 may serve as a reminder. The energy  $E_\lambda$  emitted by a black body

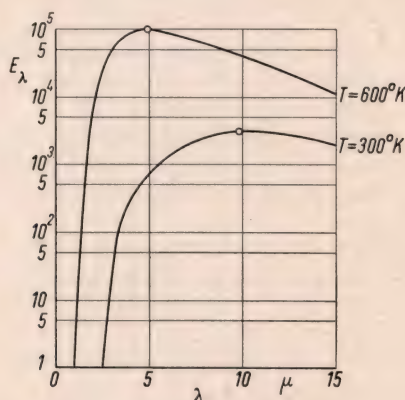


Fig. 1. Black body curves

- Planck's law,  
 ○ Wien's displacement law,  
 Area under Planck's curve equivalent to total energy (Stefan-Boltzmann's law)

at the absolute temperature  $T$  and at one particular wavelength  $\lambda$  follows PLANCK's law:

$$(1) \quad E_\lambda = \frac{\text{const}}{\lambda^5 (e^{\text{const}/\lambda T} - 1)}.$$

Integrating over all wavelengths results in the total energy  $E$  emitted by a black body. This energy is represented by the area under PLANCK's curve and is given by the STEFAN-BOLTZMANN equation

$$(2) \quad E = \text{const} (T^4 - T_1^4),$$

where  $T_1$  is the absolute temperature of the surroundings. WIEN's displacement law gives the wavelength  $\lambda_{\text{max}}$  at which the maximum radiation for a black body occurs; it reads

$$(3) \quad \lambda_{\max} = \text{const}/T.$$

It should be noted that the total energy emitted increases with the fourth power of the absolute temperature and that the wavelength for maximum radiation decreases as the temperature increases.

Using the black body equations and knowing the average temperature and emissivity of targets of military interest and the size of the emitting area, calculations have been made which show that the energy emitted by certain targets, for instance by aeroplanes, ships, certain ground installations, etc., will be sufficient for detection to be possible at an interesting range.

Since aeroplane targets are of special interest for the use of an infrared homing device, special attention was given to their emission.

The infrared radiation, its spectral intensity and its spatial distribution as emitted by an aeroplane depends on the number, power, and type of engines, the aeroplane design, the location of the engines, and the exhaust pipe installation. The main sources of infrared radiation emitted by an aeroplane are:

- (a) the hot parts of the engine, especially the exhaust pipe, and
- (b) the exhaust gases.

It was found by practical measurements that the radiation intensity of the exhaust gases of a reciprocating engine decreases very rapidly as soon as the gases have left the exhaust manifold because of the cooling effect of the atmosphere. Therefore, only the gases just behind the exhaust pipe exit need to be considered as emitters for our purposes. Near the end of the war, it was found that this is not true for jet, or rocket-propelled aeroplanes.

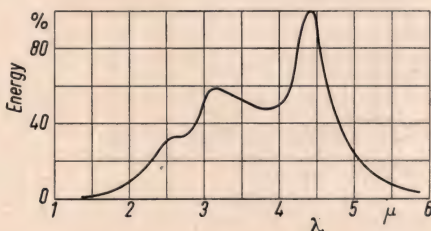


Fig. 2. Spectral distribution of infrared radiation emitted by an aeroplane

Together with the late Dr. PLUMEYER, I have made radiation measurements on various aeroplanes. As an example, Fig. 2 shows the spectral distribution of the infrared energy emitted by the end of the exhaust pipe of a reciprocating aero-engine, the BMW-6. This curve shows clearly that the spectral distribution is not equal to black or grey body radiation, but that some maxima and minima are present. We assumed that these maxima and minima are caused by the emission and absorption of gases, especially  $\text{CO}_2$  and  $\text{H}_2\text{O}$ , present in the exhaust gases and in the atmosphere. An investigation of the radiation of the hot metal pipe alone showed a distribution very similar to a grey body.

For the pattern measurements, typical results of which are shown in Figs. 3 and 4, it was assumed that an aeroplane is a point source not having the same radiation intensity in all directions. Fig. 3 represents an example of a horizontal pattern. The radius vector is proportional to the infrared radiation intensity for

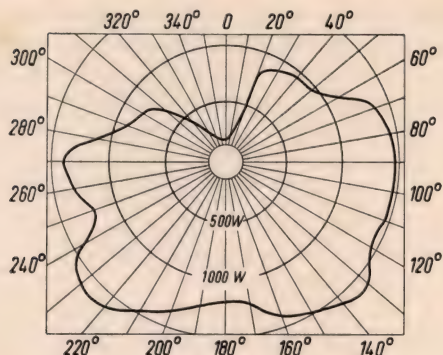


Fig. 3. Horizontal pattern of infrared radiation emitted by an aeroplane

wavelengths between 1 and approximately  $3\mu$  as emitted by a W-34 aeroplane. It shows the radiation output which a point source must have in order to emit the same amount of energy as the aeroplane used in this example, considering a horizontal plane through the aeroplane.

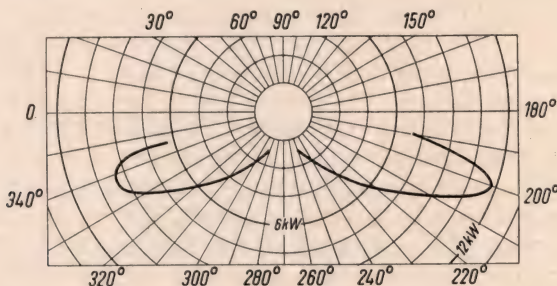


Fig. 4. Vertical pattern of infrared radiation emitted by an aeroplane

Fig. 4 shows the vertical pattern of a Ju-52. These measurements were made only in the hemisphere below the aeroplane and in a vertical plane which is fixed by the longitudinal axis of the aircraft. Similar measurements were made using other aeroplanes and also ships. These investigations were very helpful for the design of infrared detecting, tracking, or homing devices.

#### 4. ATTENUATION OF INFRARED RADIATION

After this brief review of the main problems regarding the radiation of infrared targets, we will discuss in short the influence of matter between the radiation sources and the receiver surface.

The transmission characteristics of various media and materials to infrared radiation are of considerable importance, not only because a series of such media and materials may be present between the radiation source and the receiver, so that the amount of infrared energy reaching the receiver is determined by the transmission of the intervening materials, but also because

the spectral transmission characteristics of various materials could be exploited for design of infrared filters, reflectors, or refractive elements.

The most important medium we considered was the atmosphere. We found that the transmissivity of the atmosphere for infrared radiation is greater than that for visible light. This is true under all weather conditions, even with fog or cloud. However, in cases where the transmissivity for visible light (and therefore, the range of an optical instrument) is very small, the practical absolute range of an infrared device is also small, even if the superiority of atmospheric transmission for infrared against visible light were to increase the range several times, which is very doubtful.

Considering an infrared source with an energy density at unit distance from the source of  $I_0$ , the energy density  $I_r$  at the distance  $R$  from the source may be considered as approximately equal to

$$(4) \quad I_r = (I_0/R^2) e^{-(\alpha + \beta) R}.$$

The term  $R^2$  represents the geometrical decrease of the energy. The term  $\alpha + \beta$  is the attenuation coefficient, composed of  $\alpha$  which takes into consideration the absorption in the atmosphere and  $\beta$  which takes into consideration the scattering in the atmosphere. The absorption of the atmosphere is caused specifically by the molecules of two gases, namely water vapour and carbon dioxide. We have made spectral measurements using an atmospheric pathlength of approximately 1 km and found principal transmission bands-sometimes called atmospheric windows-around 1.1, 1.25, 1.7, 2.25, 3.75, 4.7  $\mu$  and between 8 and 14  $\mu$ . The percentage of absorption depends on the amount of water vapor and carbon dioxide within the atmospheric path. Also the width of the absorption bands depends on the amount of the absorbents.

The term  $\beta$  in Eq. (4) takes care of the attenuation caused by scattering. This exponent, however, is not a constant, but a complicated function involving the wavelength of the radiation and the diameter of the particles suspended in the atmosphere and responsible for scattering. Only in the special case where the particles are all of the same size and their diameter is small in comparison to the wavelength is  $\beta$  a simple function of the wavelength. For this case, RAYLEIGH'S scattering law is valid and the scattering coefficient  $\beta$  varies inversely as the fourth power of the wavelength. Accordingly, scattering decreases considerably if the wavelength increases. These facts were confirmed by practical measurements which proved that, considering only scattering, infrared has much better transmission characteristics than visible light for fine haze, dust, or smoke, because the particles producing these effects have mostly an average diameter of less than 1  $\mu$ , which to a first approximation is small compared to wavelengths between 1 and 12  $\mu$ . For fog and clouds, infrared does not possess such great superiority because the particles are much larger than the available wavelengths. This results in a very high attenuation of infrared by clouds, heavy fogs, etc. The matter of transmission of a clear and a hazy atmosphere for infrared has been treated comprehensively by Dr. GÄRTNER.

## 5. THE INFRARED BACKGROUND RADIATION

Before discussing a few main characteristics of various infrared detector elements, a very important factor in the use of infrared for military applications

must be mentioned. This is the influence of the so-called background radiation. All backgrounds send out an infrared radiation, the total energy and spectral distribution of which depend on the temperature and nature of the background. Depending on whether the infrared receiver is directed at the sky, earth, or water, so will the background radiation be of a different nature and strength. Since the background radiation fluctuates in time and space, the receiver will indicate random noise or in some cases false targets.

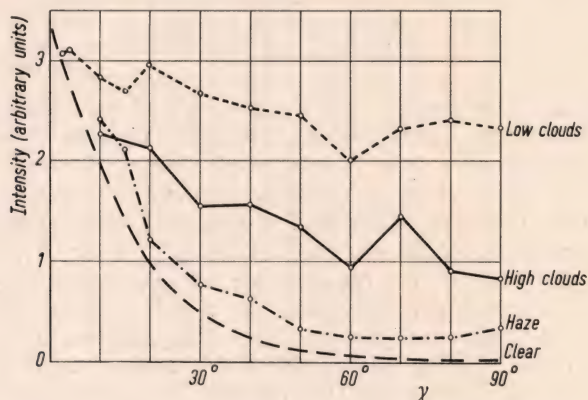


Fig. 5. Infrared sky radiation versus angle of elevation

If the interfering background radiation were constant in intensity and spectral distribution it would be very easy to eliminate its effect on the infrared device. For instance, it would be possible to produce an electric counter-current which is of the same magnitude as that produced in the receiver by the background radiation. There are many other simple methods of eliminating such a constant background radiation, but unfortunately, the background radiation fluctuates very strongly. Figs. 5 and 6 show some examples of sky background radiation. The studies resulting in these curves were based on night

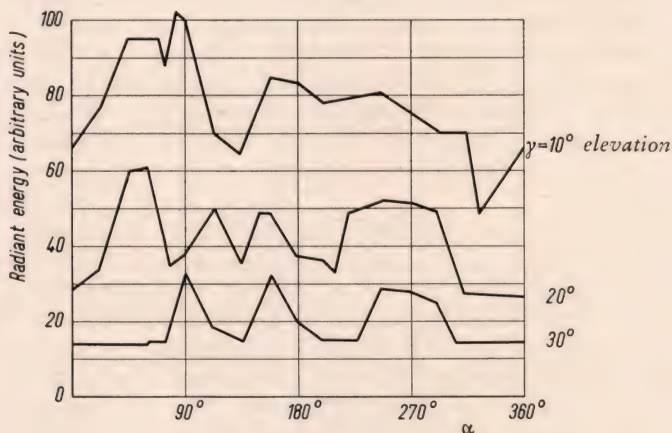


Fig. 6. Infrared sky radiation versus angle of horizon

measurements carried out during the war in Northern Germany by my assistant, Dr. HEITMÜLLER. The results were plotted as a function of azimuth and angle of elevation. For a practical device, it is very important to avoid the noise and especially false targets caused by the background radiation. In our developments before and during the war, we used filters to improve the signal-to-noise ratio. We also used special scanning methods and electronic means for discriminating background noise and target radiation.

## 6. INFRARED DETECTORS

Our next point of interest will be the main characteristics of infrared detectors suitable as receiver elements for military devices. Like the other subjects mentioned so far, the receiver problem would certainly need one or more special lectures in order to cover the physics and techniques involved in this component of an infrared device. Therefore, only a brief summary on what we did before and during the war can be given here. We considered the use of bolometers, thermopiles, and various kinds of photocells to find the most suitable receiver for a given application. The total and spectral sensitivity, the wavelength limits, the time constant, and other characteristics of such an infrared detector must be investigated and evaluated with respect to the application in a particular infrared device and to a special military problem. Thermopiles and bolometers are based on a change in temperature which is caused by the irradiation. In other words, this effect is a mass effect and there is no reason for a difference in spectral sensitivity whenever the radiation is absorbed by the sensitive material. On the other hand, this effect must be effected with a relatively great response time. Contrary to these heat machines, photocells are based not on a mass effect, but on interaction between photons and electrons, which results in a much smaller time constant and on the other hand, in the fact that photocells are sensitive only in a more or less limited wavelength region.

Searching for a photocell, sensitive to wavelengths up to at least 3, 4, or better  $5\mu$ , we rediscovered the photoconductivity of lead sulphide in 1932. We also developed photovoltaic lead sulphide cells.

Practically all infrared devices developed before and during the war in Germany, and especially the devices which were used for detection of or homing on aircraft, used as receiver elements these photoconductive cells which we had developed at the ELECTROACUSTIC Co. in Kiel. Dr. GOTTFRIED SOMMER, Dr. PICK, and Mr. KURT JUNG were my main co-operators in this part of the development.

The cells which were available during the war fulfilled sufficiently the requirements of such an infrared detector. Extensive research and development work was still going on at the end of the war. The performance of a lead sulphide photoconductive cell is based on the so-called internal photoeffect which means that absorbed photons change the resistance of the material by either changing the number of electrons within the conduction band and/or creating holes in the first occupied energy band. The lead sulphide cell of the ELECTROACUSTIC Co. was produced by using a chemical precipitation method. Cells with various sizes and shapes of the sensitive area have been made.

A large scale production was under way. Similar cells were developed by Dr. GÖRLICH at the laboratories of ZEISS-IKON in Dresden. GÖRLICH refined and used an evaporation method which was developed by the late Prof. GUDDEN.

Much time and effort was spent in continuously improving the paramount properties of both types of cells, especially by modifying the production parameters and introducing the right impurities in the optimum amount.

The sensitivity of those cells could be increased by a factor of 10 to 20 by cooling them. Liquid air and later solid  $\text{CO}_2$  were used as coolants.

In 1944 the first lead selenide photoconductive cells were ready in our laboratory for comparison measurements. It was proved by field tests that due to the shift of the upper wavelength limit toward longer wavelengths these cells gave at least the same practical ranges in spite of the fact that their sensitivity at the spectral peak was approximately one order of magnitude smaller than that of the lead sulphide cells.

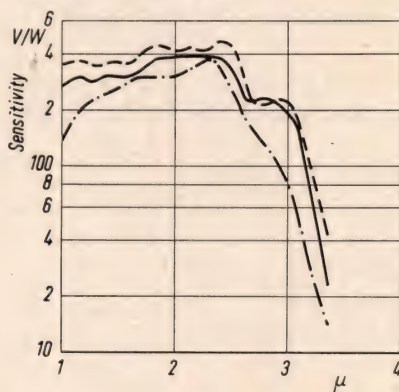


Fig. 7. Spectral sensitivity of lead sulphide cells

The time constant of both the lead sulphide and the lead selenide cells at dry ice temperature was in the region of one tenth of one millisecond. The upper wavelengths limit for the PbS cell was approximately  $3\mu$  and for the PbSe cell approximately  $4.5\mu$ . The sensitivity in terms of radiation necessary to produce a signal-to-noise ratio of 1 was for a good PbS cell of the order of  $10^{-10}$  to  $10^{-11}$  watts for a black body at the temperature of  $500^\circ\text{K}$ , a modulation frequency of 100 c/sec, and a bandwidth of 1 c/sec. Fig. 7 shows the spectral sensitivity of typical PbS cells as measured in the early 1940's.

## 7. OPTICAL SYSTEMS

Another very important component of an infrared device is the optical system. We used mirror systems as well as lens systems. Unfortunately, for the wavelength region in question, not enough special materials having a sufficient variety of index of refraction were available to permit a good optical design. It should be mentioned that an excellent material was developed by ZEISS under the name KRS 5. This material was a mixture of thallium iodine, and thallium bromide; it had a very small absorption in the near infrared. The

index of refraction was relatively high, resulting in high reflection losses. However, an anti-reflection coating was developed and it was possible to optimize for a particular wavelength band. Special glasses we used were the Duran glass, manufactured by SCHOTT Glass Works, and a very promising material which was developed by Prof. KLIEFOTH. The later material was a so-called aluminum glass. HERAEUS developed a fused quartz which had almost no absorption at the water band around  $2.6\mu$ . Various mirrors and mirror systems were developed and used. Spherical aberration and coma were corrected by the use of SCHMIDT plates or correcting mirrors as designed, for instance, by Prof. PICHT. By using such a corrected optical system, it was possible to increase the width of the field of view over which sharp images could be obtained, and in addition it was possible to obtain a high optical power. We used optical systems with an  $f$ -number down to approximately 0.6.

## 8. RANGE AND ACCURACY

Summarizing our discussion we may now list the parameters which define the range and accuracy of an infrared device. The range parameters are the following:

- the infrared energy emitted by the target,
- the energy emitted by the background and the efficiency of means for discriminating between target and background radiation,
- the atmospheric attenuation,
- the diameter of the optical system and its efficiency,
- the sensitivity of the detector element and its time constant,
- the field of view,
- the desired resolution, and
- the frame time.

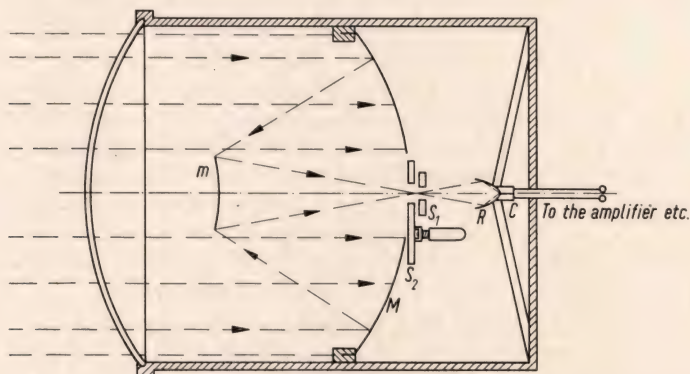
The accuracy is mostly determined by the geometry of scanning and the image quality of the objective lens or the optical system used. The practical infrared homing devices developed during the war had a range of approximately 5 km when used against a bomber with three piston engines flying under fair weather conditions at a medium altitude. The angular accuracy was in the order of one tenth of a degree and the time constant for the complete system was for most of the devices of the order of a tenth of a second or smaller.

## 9. REMARKS ABOUT INFRARED HOMING DEVICES

In considering an infrared homing device for tactical use, it is important to bear in mind that such a device has a relatively small range, usually smaller than that of a radar homing device. However, an infrared device is capable of the high accuracy inherent in optical instruments. These characteristics suggest that such a device may be of particular value for the terminal phase of the flight path of a missile. Of course, the method appeared to us very promising for cases where the missile was launched at short distances against a target which had already been detected by the infrared homing device.

The philosophy of our development was to guide and home a missile into collision with or at least very close to the target. If the missile is fired from a relatively safe distance by using a homing device, the pilot has no operation limitations in flying his aeroplane after firing. The ultimate design of an infrared homing device has to meet all the general technical requirements pertinent for an aircraft-borne device. As far as the special design of the device as an infrared instrument is concerned, we found the great importance of tailoring the device as early as possible to the missile and its controls in which the device will be a component or a subsystem. The missile with all its parts should be designed as a whole, or, in other words, we should develop and design a functional and reliable "system". This last objective was not accomplished at the end of the war.

The proper design of an infrared homing device depends to a great extent upon the tactical employment of the missile and the characteristics of its control system. We considered the use of an infrared homing device for all types of missiles for ground to ground, ground to air, air to ground, and especially for air to air applications.



*Fig. 8. Fundamental structure of an infrared homing device*

Fig. 8 shows the fundamental structure of an infrared homing device. The infrared energy emitted by a target is gathered by an optical system and focused in a certain area of the focal plane, the size of which depends on the focal length of the optical system, and the desired field of view of the device. After scanning this field, the infrared energy is then concentrated on to the infrared detection element by using optical components. The output of the photocell is amplified and the final intelligence must be transferred in a usable manner to the missile control.

With respect to the angular area of operation, which is the solid angle within which the homing device is required to detect and track a target, the following three factors are of special importance:

- (a) The angular size of the area within which the homing device is required to detect the target,
- (b) the method of scanning, and
- (c) the method of fixing the position of the target image relative to the centre of the field of view and thereby defining the angular off-course position of the target with respect to the optical axis of the homing device.

There are four different angular areas to be considered.

(1) The instantaneous angular area. Normally this is a certain percentage of the image area representing the field of view from which at any given moment, radiation is being received by the detector. It is determined by the optical system and the method of scanning.

(2) The optical field of view. This is the angular area which is scanned. It is determined by the optical characteristic of the device and the area of the sensitive layer of the infrared detector.

(3) The follow-up area. This is the angular area within which the homing device can move, in order to retain the detected target continuously in the field of view.

(4) The search area. This is the field of view increased by a mechanical movement of the device or parts of it.

According to the foregoing, there are two fundamentally different types of homing devices. In one type the optical axis is fixed relative to the axis of the missile. This means there is no follow-up system and no search movement. In the other type, the optical axis changes direction in accordance with the action of the mechanical follow-up or search system, thereby either keeping the target within the field of view or searching for the target.

We found it advantageous to design a homing device in such a way that the field of view is as small as possible. This is necessary to minimize the interfering background radiation and thereby obtain greater ranges. For a smaller field of view, a smaller sensitive area of the photocell can be used, which also results in a greater sensitivity or range. The angular accuracy also increases as the size of the field of view decreases. Another advantage of a small field of view is that it decreases the possibility that the homing device will detect false targets or decoys and cause the missile to deflect from its proper course.

Of course, the size of the field of view must be adapted to the type and characteristics of the missile, the type and characteristics of the controls, the tactical conditions, and the target of interest.

The method of scanning must be adapted to the specific requirements of each missile.

For a missile requiring simple on-off control signals, the scanning of the field of view by mechanical tracking created no great difficulties. For example, a semi-circular chopper as shown in Fig. 9, will readily fulfill the specifications for a missile requiring only up-down, left-right signals. In this particular system, the position of the target in relation to the optical axis of the device was obtained in the following manner: A commutator moves synchronously with the semicircular disc. If the position of the target image is as shown in Fig. 10 for example, the alternating current represented at *A* in the figure is produced at the amplifier terminal. If a commutator is now used to constantly reverse the current after each half period the curve of potential shown at *B* is obtained. The direct current instrument or relay which lies behind the amplifier uses the direct current component and consequently shows a deflection or activates a contact. A second commutator commutates a quarter of a period earlier or later. The resultant potential is shown at *C*. A second direct current instrument or relay, in this case, shows no deflection or operation. Since the direct current component is zero, the instruments, or the relays indicate that

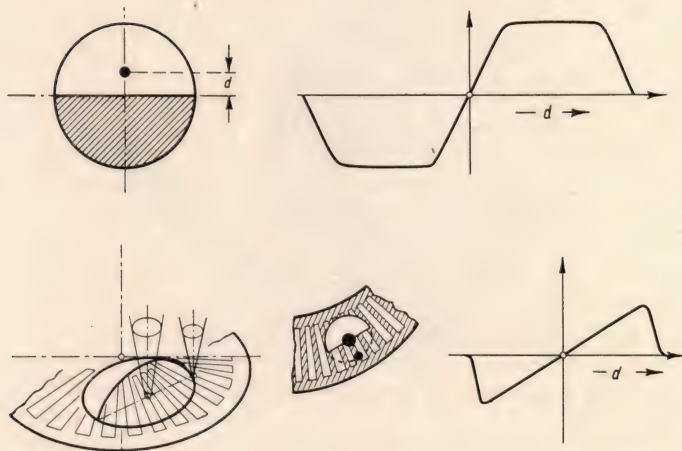
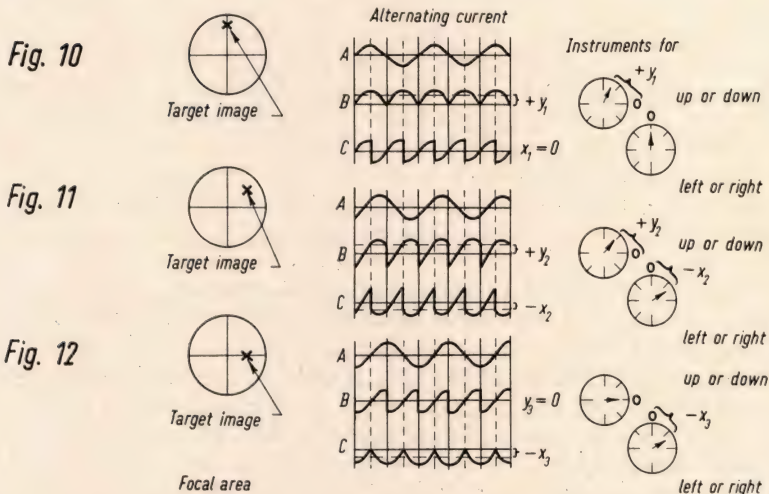


Fig. 9. Semi-circular scanning method



Figs. 10—12. Performance of a semi-circular scanning disc

there is upper deviation of the target image in relation to the centre of the modulating disc, but no deviation to the left or right. It is easy to see that with a position of the image shown in Fig. 11, both of the instruments or relays will operate. In Fig. 12, the target image is shifted to the right but not upwards or downwards. Consequently, the height indicator shows no deflection. The lateral indicator on the other hand shows a deflection to the right. A homing device developed on this principal was ready and intended for installation in the BV-143. The construction of the instrument is shown in Fig. 13.

There were missiles, however, whose guidance requirements were not satisfied by a simple on-off homing system. Their control system required error values corresponding to the off-course position of the target. The problem of

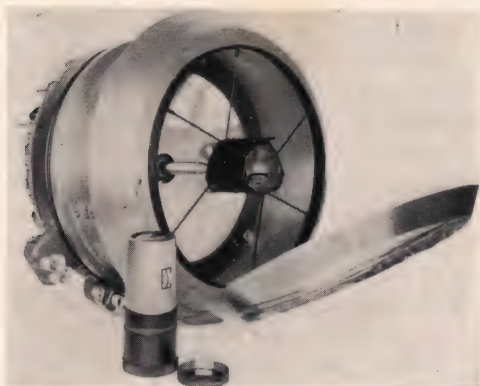


Fig. 13. Infrared homing device "Hamburg" for BV-143

providing for proportional control signals was solved, for instance, by using a spherical segment as shown in Fig. 9 or by using a polar-shaped scanner as shown in Fig. 14. In this figure it may be seen that because of the rotation of the disc,

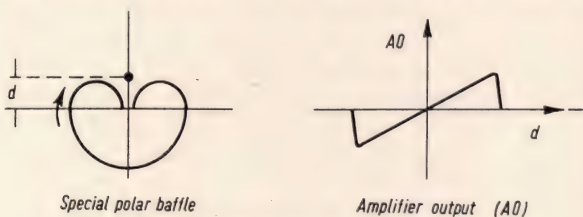


Fig. 14. Polar scanning disc

the greater the distance  $d$  of the target image from the centre of the field of view, the longer the detector is exposed to the infrared energy. The result is indicated by the output as shown in Fig. 14. At the lower part of Fig. 9 a half-spherical segment baffle together with a modulating disc is shown. Due to the shape of the half-segment and the image size in planes parallel to the focal plane, a relationship between signal strength and angle-off exists. Since the radiation energy depends on the distance between target and receiver, the amplifiers were in some cases equipped with an automatic gain control in order to obtain the same relationship between the off-course position of the target and the output of the amplifier at all times.

In the case of an infrared homing device which was not fixed to the missile, the output signal of the homing device was used to turn the homing device itself within the missile into the direction of the target. Simultaneously, electrical values indicating the extent of this movement were transferred to the control system. The actual direction of the target, however, may differ from the direction of the optical axis of the homing device by a small amount. A signal representing this deviation was added to the first signal and relayed to the controls of the missile. The accuracy of guidance was increased thereby, whereas the need for precision components in the tracking system of the homing device was not so great. The electrical signals corresponding to the mechanical movement of the tracking system were continuously corrected by

the values corresponding to the deviation from the optical axis. Homing systems based on the principles described were developed by the ELECTROACUSTIC CO. in Kiel, under the name "HAMBURG". During the course of the development certain refinements and modifications were introduced. A few of them were the following:

The original devices used the semi-circular disc described, both for scanning the field and modulating the radiation. For certain applications, an additional rotating disc with radial slots for modulating was used in addition to the scanning semi-circular disc.

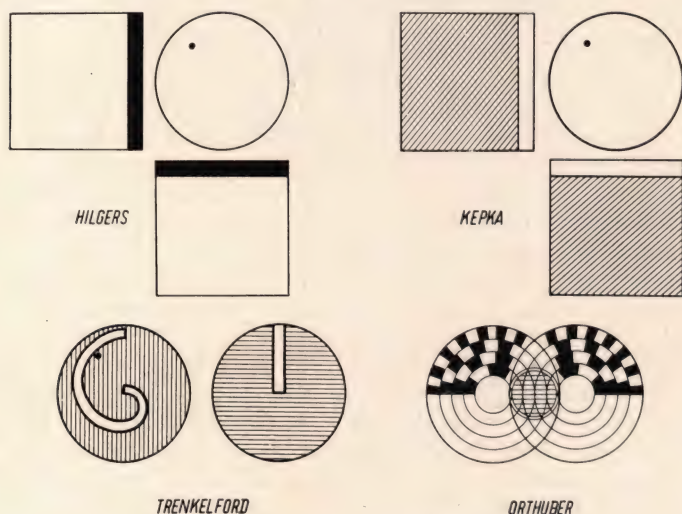
For applications in which the homing device detected the target before launching the missile, it was planned to use a narrow optical field just great enough to overcome the launching errors, so that the target was not lost and could still be seen by the homing device when the missile had stabilized, and had reached a velocity at which it responded to normal controls. An optical field of view of approximately  $\pm 3^\circ$  to  $\pm 6^\circ$  was considered sufficient. In cases where larger angles between target and missile were expected, an installation capable of an additional mechanical movement was designed. A follow-up movement up to approximately  $\pm 10^\circ$  to  $\pm 40^\circ$  was under development. For another version, a spiral movement of the instrument was developed, resulting in an angular coverage for search of approximately  $\pm 10^\circ$  and it was planned to increase this angle to approximately  $\pm 25^\circ$ . The search movement was stopped after the target was picked up. This search feature was especially necessary if the missile was fired before the homing device had located the target. For these cases the homing device was supposed to control the missile during the last phase of the flight. Depending on the flight performance of the missile and on the desired course, the search cone was either centred around the missile axis or another axis in space. A stabilized-platform installation was in preparation. In the air-to-air missile Hs 298 the position of the optical axis was given by the position of an angle-of-attack vane defining at each moment the tangent to the flight path. This resulted in a pursuit course.

In addition to the infrared homing devices we developed an infrared proximity fuse for increasing the probability of destruction. It was planned to integrate both devices into one unit. The principle of the infrared proximity fuse was as follows: A lead sulphide photocell was installed in the nose of the missile perpendicular to the missile length axis. During the missile flight, the output of the cell increased with the decreasing range to the target reaching a sharp maximum at the smallest distance between missile and target. This pulse was amplified and used to control a relay for operating the fuse. Practical tests have shown that this device worked satisfactorily at suitable ranges.

For cases where the homing device detected the target before launching the missile, the homing device signals should be displayed to the pilot. For other cases, additional infrared search devices installed in the aeroplane were developed. My main co-operators in the field of system development were Dr. RÜCKLIN, Dr. OCHMANN, Dr. AHRENS, Dr. HEITMÜLLER, and Dipl.-Ing. ORLICH.

Other infrared homing devices which were under development used in general the same principles we have discussed. The main difference between these various devices was the scanning method in order to obtain the angular information.

A device proposed by Dr. HILGERS of the AEG used a photocell with an area large enough to cover the total field of view in the focal plane. Two very narrow mechanical stops were moved perpendicular to each other across the cell. As soon as the image was covered by a stop a pulse was generated. The position of the pulse was measured with respect to an electrical phase reference generated by an AC generator, the rotation of which was synchronized with the movements of the stops. Fig. 15 shows this method schematically. The firm



*Fig. 15. Pattern of various scanning discs*

KEPKA, in Vienna, used a similar principle by moving two narrow slits across the field. Their optical system and cell size resulted in a field of view of  $\pm 1.8^\circ$ . The device was able to follow within an angle of  $\pm 100^\circ$ . It was also possible to search a field of  $\pm 20^\circ$ . The search motion of the device was powered by pneumatic air turbines.

Mr. TRENKELFORD associated with RHEINMETALL-BORSIG developed a system for which the scanning was accomplished by two rotating discs, one of them having a spiral opening, the other one a slit. The position of the image within the focal plane was defined by the position of the two scanning discs relative to each other.

Dr. ORTHUBER of the AEG used a frequency modulation for scanning. The position of the target within the focal plane was given by a frequency, generated by two discs scanning the field of view. Both discs had a few circles of radial slots, the number of slots per degree being varied from the centre to the edge, and overlapped each other, resulting in a different modulation frequency for the various areas within the field of view.

In conclusion, it should be noted that after considering the advantages and disadvantages of an infrared method and the results of the laboratory and field tests carried out on experimental devices, the people concerned were convinced in 1944 and 1945 that an infrared homing device constituted an excellent solution to certain guidance problems for missiles.

## DISCUSSION

Dipl.-Ing. F. MÜNSTER (Düsseldorf): You mentioned infrared measurements carried out on various aeroplanes having reciprocating engines. In this case you found that the hot metal parts of the engine, especially the exhaust pipe or exhaust stack, constituted the main source of the infrared radiation, whereas compared with this radiation the emission of the exhaust gases is negligible a short distance behind the exhaust pipe opening. You mentioned also that the amount of infrared radiation emitted by the exhaust of a jet or a rocket engine is much greater than in the first case.

It this remark based on comprehensive measurements?

Dr. E. W. KUTZSCHER: During the War we made a few measurements on the total infrared emission of rockets and also jet engines. In both cases it was found that the total exhaust gas radiation is much greater than the emission of the exhaust gas of a reciprocating engine. Results of recent measurements are classified.

Prof. Dr. QUICK (Aachen): Dr. KUTZSCHER, you mentioned measurements about the spectral absorption of the atmosphere. It was interesting to note that the curve of the spectral measurements has some peaks and dips and that spectral regions exist for which the atmospheric absorption approaches zero.

Would you please say some more words about the physical reasons for this phenomenon?

My second question concerns the change of atmospheric absorption with altitude. I assume that the attenuation of infrared radiation decreases with increasing altitude.

Dr. KUTZSCHER: We obtained results from various spectral absorption measurements. We used the infrared radiation of a given source and measured its spectral absorption within the clear atmosphere using various pathlengths. The spectral resolution used was only of the order of a tenth of one micron; however, we found "atmospheric windows" within the infrared region investigated. These windows correspond to regions in which practically no water vapour or carbon dioxide absorption bands exist. The absorption by these gases is caused by the characteristic oscillations of their molecules. The atmospheric absorption within the infrared region depends mostly on the total water vapour and carbon dioxide content within the air mass through which the infrared radiation has to travel. These total amounts of  $H_2O$  and  $CO_2$  gases influence both the percentage of absorption at a given wavelength and also the spectral width of the band. Since water vapour and carbon dioxide content normally decreases with increasing altitude, the atmospheric absorption of infrared radiation becomes much smaller at greater altitudes. In addition, scattering by condensed water will also be smaller at high altitudes, resulting in a small overall attenuation of infrared radiation.

# REVIEW OF THE DEVELOPMENT OF PROXIMITY FUSES

FRIEDRICH VON RAUTENFELD \*

## 1. THE PROBLEM AND ITS REQUIREMENTS

In standard "classical" missiles, the explosive charge is detonated by a pre-set time fuse or by a percussion fuse. With guided or "homing" missiles, such detonation methods are not feasible, since the flight time of such missiles can not be pre-determined and, furthermore, it would be extremely difficult and expensive to improve the flight control to such an extent that a direct hit would be obtained with absolute certainty <sup>1</sup>. The problem involved, therefore, is as follows: to initiate ignition automatically by an effect caused by the target itself. Thus, the "effective range" of the missile will be increased so that a "virtual" hit will result even in spite of inaccurate flight control.

The distance of the missile from the target represents an obvious criterion for automatic ignition. Fig. 1 shows on the left (part I) schematically that ignition is initiated automatically when the distance from the target becomes less than

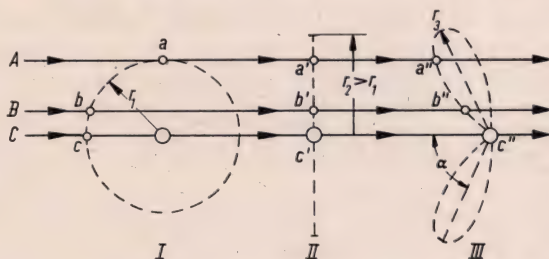


Fig. 1. Principles of proximity fuses

I = Distance fuse

$r_1$  = Initiating distance

$a, b, c$  = Ignition points at distance  $r_1$

$A, B, C$  = Flight paths

II = Approach fuse

$r_2$  = Maximum response distance

$a', b', c'$  = Ignition points at minimum distance in by-pass flight

III = Directional fuse

$r_3$  = Range of the diagram

$a'', b'', c''$  = Ignition points at entering the diagram

$\alpha$  = Half aperture angle of the directional cone

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$r_1$ . Therefore, the directional diagram of the idealized distance fuse is a sphere of radius  $r_1$ . This radius, of course, must be so adjusted to the explosive force of the missile that the latter will be detonated only when passing close enough to the target (e. g. flight path *A*, detonation at *a*). This illustration, though, reveals a disadvantage of the distance fuse: if the missile is guided accurately (flight path *C*), a direct hit will be prevented by the proximity fuse responding prematurely (at point *c*). Thus, a part of the explosive force will be spent ineffectively. The choice of the correct initiating distance  $r_1$  has to take these two contradictory viewpoints into account. On the one hand,  $r_1$  should be as small as possible to make sure that the full effect of the missile is obtained; on the other hand,  $r_1$  should be so large that at least some effect will be obtained even if the target is missed.

Apparently, fuse II will be more effective, which — as shown in Fig. 1 (centre part) — will respond when the missile is passing at a minimum distance from the target. For the flight paths *A* and *B* ignition will invariably occur at the points *a'* and *b'*, for flight path *C*, however, not prior to reaching point *c'*, i. e. in a direct hit. Devices working on this principle have more aptly been named "approach fuses"<sup>1</sup>. The directional diagram of such a fuse is a circular plane of radius  $r_2$ . It is obvious that even for a large response distance  $r_2$  no explosive force would be wasted ineffectively. Therefore the choice of the operating radius  $r_2$  is rather un-decisive. Generally, radius  $r_2$  for the approach fuse may be larger than radius  $r_1$  for the proximity fuse.

As will be easily understood, the functioning of an approach fuse is more complicated and requires a larger amount of technical components than a distance initiated fuse. It is possible, however, to obtain a performance similar to that of an approach fuse by giving the device a certain directional effect: "Directional fuse". As emerges from Fig. 1 (part III), the directional diagram approaches the form of a cone with an aperture-angle of  $2\alpha$ . The fuse will work as soon as the target enters the diagram (at *a''* or at *b''* or — by direct hit — at *c''*). When  $\alpha = 90^\circ$ , the function of an approach fuse according to Fig. 1 (part II) will be obtained. Nevertheless, the term "directional fuse" will still be appropriate, since the working mechanism differs physically from that of an approach fuse.

The effective range of proximity fuses or approach fuses should largely depend upon the explosive force of the missiles. Generally, ranges from 5 to about 50 m are required. Fortunately, these effective range requirements are much easier to comply with than those for homing devices, where working ranges around 1 km are required<sup>2,3</sup>.

An important point in the design of proximity fuses is the problem of simple construction and production, because the device is completely lost in action. Therefore, quite a lot of physically possible designs cannot be considered. Other self-evident requirements, such as reduced volume (about 1 litre), low weight (about a few kilograms), satisfactory resistance to vibration, aerodynamically smooth aeriels etc., will not be discussed here.

## 2. PHYSICALLY FEASIBLE SOLUTIONS

As in the case of homing devices, the possible types of automatic fuses may be classified by the nature of the energy used and subdivided into active and

passive systems. Electronics offers numerous useful possibilities, but fuses have also been built which rely on electrostatic, magnetic, optical and acoustical phenomena.

It will first be explained how the most advantageous function of an "approach fuse" may be obtained by simple circuits.

Fig. 2 (part I) above shows e.g. the time-chart of the input voltage  $U_i$  induced in the system by some effect when passing the target in a lateral distance  $a$  short or great. Let an evaluation voltage  $U_1$  be proportional to  $U_i$ . When the ignition is initiated by exceeding a pre-adjusted minimum value  $U_{1\min}$ , the detonation point may be reached at different distances before the missile reaches its minimum distance from the target ( $d_1$  or  $d_2$ ).

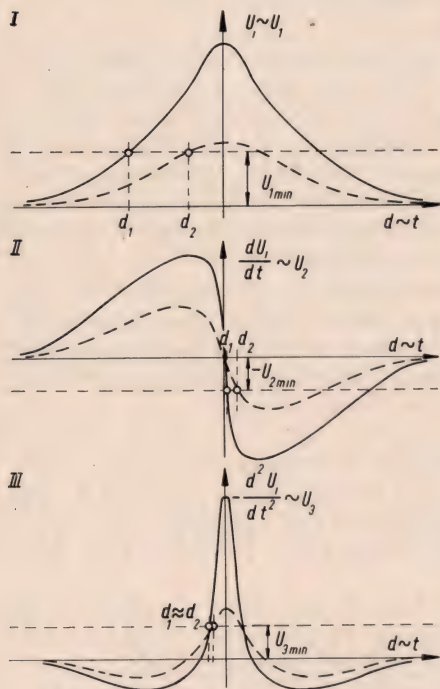


Fig. 2. Time-chart of input voltage when passing the target

$U_i$  = Input voltage  
 $U_1$  = Evaluation voltage  
 $d_{1,2}$  = Distance from the target in flight direction  
 ——— Distance a short  
 - - - Distance a great

If, however, instead of the base voltage  $U_1$ , the first derivative with respect to time  $U_2 = dU_i/dt$  is used to initiate ignition — Fig. 2 (part II) —, almost an approach fuse is obtained and, the higher the sensitivity of the electrical system ( $U_{2\min}$  as small as possible), the more is this so. In practice such differentiation with respect to time can be easily accomplished, e.g. by using an RC circuit of a small time constant as shown in Fig. 3. An unintentional detonation (e.g. one caused by jamming) can be avoided by connecting the two relays  $R_1$  and  $R_2$  in series (Fig. 3), the base voltage  $U_1$  first priming the fuse, then — with the change of sign of  $dU_i/dt$  — following the actual ignition initiation. Finally, it should be mentioned that a still more accurate adjustment of the detonation point can be realized by using the second derivative with respect to time of the input voltage  $U_i$ , as shown in Fig. 2 (part III). For this

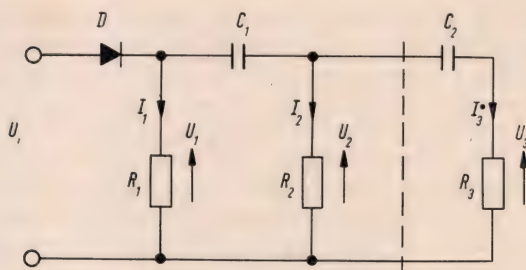


Fig. 3. Differential circuit

$I_{1, 2, 3}$  = Currents  
 $R_{1, 2, 3}$  = Resistances (relays)  
 $D$  = Diode  
 $U_{1, 2, 3}$  = Voltages  
 $C_{1, 2}$  = Condensers

purpose, the differential circuit  $R_3 C_2$  is provided in Fig. 3. Since the peak value of the second derivative  $d^2 U_i / d t^2$  depends upon the lateral distance between target and flight path, this value can be used for a more accurate determination of the maximum response distance  $r$ .

The performance of an approach fuse can be exactly obtained by using the DOPPLER effect. This effect is rather small and may, therefore, be considered only for active (or semi-active) systems. Fig. 4 (part I) shows three flight paths  $A$ ,  $B$  and  $C$  with different lateral distances  $a$ . By reflection, a frequency shift proportional to the relative radial velocity  $v_r = v \cos \alpha$  occurs, producing a DOPPLER beat.

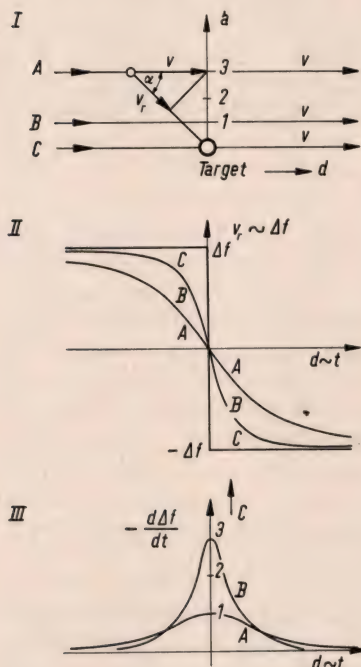


Fig. 4. Principles using Doppler effect

$v$  = Velocity of flight  
 $v_r$  = Relative radial velocity  
 $a$  = Lateral distance of target  
 $\Delta f$  = Frequency shift  
 $d$  = Distance from the target in flight direction  
 $t$  = Time

The curves in Fig. 4 (part II) show that, at the minimum distance from the target, a change of sign takes place in the DOPPLER shift (passes through zero with a phase shift) which may be used for initiating the fuse-ignition. The time derivative of the DOPPLER shift  $\Delta f$ , as shown in the curves of Fig. 4 (part III) is — assuming a constant velocity  $v$  of flight — strictly inversely proportional to the lateral distance  $a$  of the target from the flight path whenever  $v_r$  passes through zero. The peak value of  $d v_r / d t$  can thus be used for the exact definition of the maximum response distance of the fuse, regardless of the reflective properties of the target and of sensitivity variations in the transmitting and receiving equipment.

### 3. EXAMPLES OF GERMAN DEVELOPMENTS

In the following, a number of devices developed and tested during the last war will be discussed. Today, it is hardly possible to give a complete review containing reliable numerical data. However, the systems which are described below should give a good over-all impression of the work done.

#### 3.1. Electronic Fuses

To begin with, the high frequency electrical reflection systems will be discussed systematically according to the wavelength employed. It should be stressed here that, due to the required close resolution, only *pulse-free* methods can be considered<sup>4</sup>. The technical production of sufficiently short pulses would be very difficult. In addition, the fact that the technical expenditure on the receiver depends more on the close resolution than on the maximum range, is often overlooked. So, for example, if the minimum distance is to be halved, the receiver bandwidth has to be doubled. Since the product of bandwidth and amplification is constant for each stage, it also follows that the number of stages has to be doubled.

##### 3.1.1. "Trichter" (Blaupunkt-Werke, Berlin)

The automatic fuse "TRICHTER", built by the BLAUPUNKT-WERKE, Berlin (responsible engineer: Dr.-Ing. G. GÜLLNER), was envisaged as supplementary equipment to the homing device "MAX-A"<sup>2</sup>. The set used the wavelength  $\lambda = 3.9$  cm, produced by the magnetron transmitter of the "MAX-A". Fig. 5 shows the lay-out of the device. A small part of the transmitter output was conducted over a coupling resistance  $R_c$  (which is essential for the operation of this system) to an antenna of four co-axial half-wave elements. The same antenna was used to receive the reflections from the target.

The axially-symmetrical directional diagram of this antenna had the shape of a flat circular disc. Thus, a "directional fuse" was obtained, which in operation corresponded closely to the idealized approach fuse, especially as the antenna diagram appeared twice (as transmitter and as receiver). The received input power, shifted by the DOPPLER frequency of a few kc/sec, was now picked up by the receiver. The coupling resistance  $R_c$  prevented the receiver input from being entirely "absorbed" by the transmitter and from being consumed in the source resistance  $R_s$ . The entry circuit of the receiver consisted

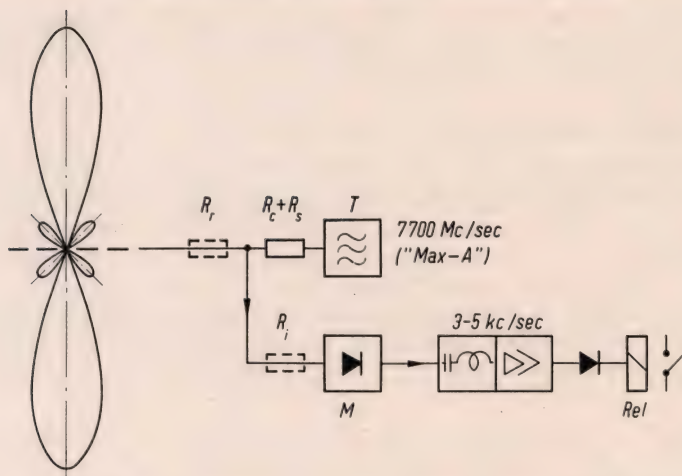


Fig. 5. Lay-out of automatic fuse "Trichter"

 $T$  = Transmitter $R_s$  = Source resistance of  $T$  $R_r$  = Radiation resistance of the antenna $Rel$  = Relay $R_c$  = Coupling resistance $R_i$  = Input resistance of  $M$  $M$  = Mixer stage

of a mixer stage  $M$  which, besides the reflected power, picked up a part of the transmitter output (without frequency shift) as oscillator frequency. In this way, the input could be converted to the low DOPPLER frequency in the range between 2 and 5 kc/sec and then selectively amplified. Thus, with simultaneous transmission and reception (without the use of pulses) and with simultaneous use of a joint antenna, a frequency separation of the primary and of the reflected radiation was achieved in a physically elegant manner. The technical difficulty of this method was the fact that the primary frequency, due to noise modulation, was also apt to contain frequency components in the range of the reflection frequencies. This becomes evident on considering the high frequency amplitude modulation occurring at the input terminals of the mixer stage  $M$ . The receiver input power  $N_r$  becomes <sup>4</sup>, assuming

$$R_c + R_s = R_r \ll R_i$$

by mirror-like reflection by a target which is very large compared with  $\lambda$ :

$$(1) \quad N_r = 10^{-2} (\lambda/r)^2 G_t G_r N_t.$$

Here  $G_t = G_r = G$  are the "power gain factor" of the transmitter and receiver antenna respectively, and  $N_t$  is the output power of the transmitter. With the values

$$\lambda = 3.9 \text{ cm},$$

$$G = 2.9 \text{ (four co-axial half-wave dipoles),}$$

$$r = 20 \text{ m},$$

one obtains the degree of useful modulation

$$(2) \quad m = \sqrt{2 N_r / N_t} = G \lambda / r \sqrt{2 \cdot 10^{-3}} \approx 0.8 \cdot 10^{-3}.$$

Nevertheless, by selective amplification a sufficient ratio between effective input and noise power could be achieved. As far as is known, however, developments were not wholly completed.

### 3.1.2. "Kakadu" (Donag, Vienna)

The fuse "KAKADU" was developed by the DONAG, DONAULÄNDISCHE APPARATEBAU-GESELLSCHAFT, Vienna (responsible engineers: Dr. J. SCHWARZMANN and Dipl.-Ing F. BEDENIG, assisted by Prof. BENZ). The device used a wavelength of  $\lambda = 0.5$  m. Both transmitter and receiver were equipped with a half-wave dipole (Fig. 6). Both antennae  $A_t$  and  $A_r$  were arranged co-axially and were decoupled by quarter-wave wave-traps and partly by a shielding ring so that the receiver could respond to the reflections from the target. The transmitter consisted of an RL 12 T 1 valve in a three-point circuit, the emitted

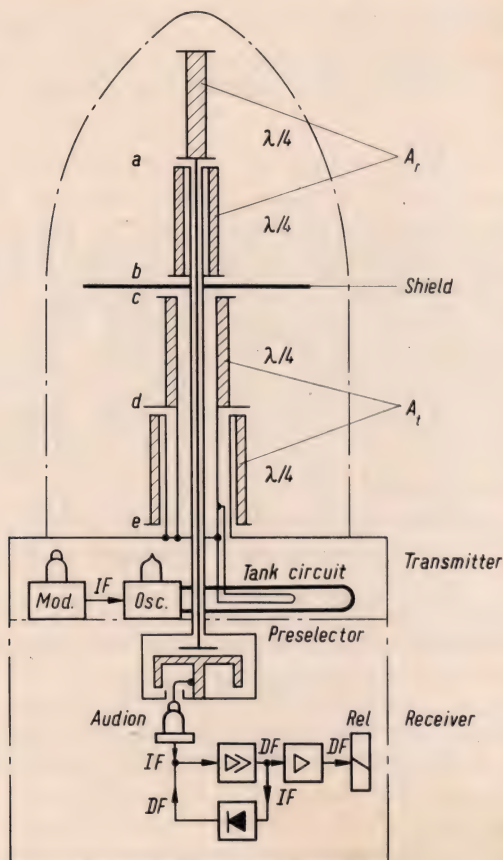


Fig. 6. Lay-out of automatic fuse "Kakadu"

$A_{t,r}$  = Antenna of transmitter and receiver respectively

$a \dots e$  = Adjusting rings

Mod. = Modulator stage

Osc. = Oscillator

IF = Intermediate frequency DF = Doppler frequency

power amounting to 0.1 W only. The receiver consisted of an input stage in an audio circuit connected in series with an amplifier in a reflex circuit which was selectively tuned to a frequency of 20 c/sec. As soon as the DOPPLER beat had dropped to this value whilst the missile approached the target, the fuse relay at the output terminals of the receiver was set to work. The maximum controlled distance was about 15 m. By correctly tuning the adjusting rings *a ... e* (Fig. 6) the transmitter-receiver-coupling could be reduced to the small value of  $C_{rt} = 10^{-2}$ . From the equation

$$(3) \quad N_r = 4 \cdot 10^{-2} (\lambda/r)^2 N_t$$

for the receiver input  $N_r$ , assuming  $C_{rt} = 10^{-2}$ ,  $\lambda = 0.5$  m and  $r = 15$  m, the degree of modulation  $m$  at the input of the receiver is found to be <sup>4</sup>

$$(4) \quad m = (1/C_{rt}) \sqrt{N_r/N_t} = (0.2/C_{rt}) (\lambda/r) \approx 0.6.$$

Using a total of 5 valves (4 of which were of the type RV 12 P 2000), the device weighed about 7 kg. It fitted into a cylindrical space 100 mm in diameter and 150 mm in height. The device was ready for production and an order for 20,000 units or so had been placed. It was intended to fit the "KAKADU" mainly into the missile Hs 293. The principal difficulty was the critical adjustment of the antennas, in particular with the regard to a necessary 20 g vibration insensitivity.

### 3.1.3. "Marabu" (Siemens & Halske, Berlin)

The "MARABU" fuse built by the Central Laboratory of SIEMENS & HALSKE, Berlin (responsible engineers Dr. E. ALSLEBEN, Prof. H. SCHÖNFELD and Obering. H. SCHUCHMANN), worked on the principle of the SIEMENS radio altimeter FuG 101 a <sup>5</sup> with frequency modulation.

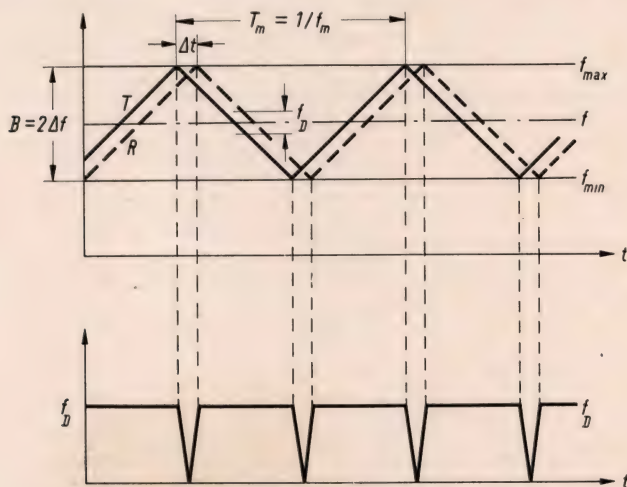


Fig. 7. "Marabu" principle of frequency modulation

$T$  = Transmitter frequency radiation     $R$  = Reflector frequency radiation  
 $f, f_{min}, f_{max}$  = Frequencies  
 $B = 2\Delta f$  = FM deviation     $t$  = Time  
 $f_D$  = Frequency difference

This principle, used in various forms in altimeters <sup>6,7,8</sup>, is explained in Fig. 7: the frequency  $T$  of the transmitter is modulated by means of a motor-driven condenser causing a deviation  $B = 2\Delta f$  in accordance with a linear time-function ("Saw tooth"). This frequency returns via the detour transmitter — target (ground) — receiver as shown in curve  $R$ , i. e. with a time-shift  $\Delta t = 2r/c$ .

It follows that there is a frequency difference  $f_D$  between the primary radiation  $T$  and the reflection  $R$ , which is proportional to the height  $r$ :

$$(5) \quad f_D = 4 f_m B r/c.$$

The interference of the two radiations in the receiver supplies the frequency difference  $f_D$  which, in the altimeter, may be changed directly into an indication calibrated in metres (up to 1000 m).

The "MARABU" worked on a wavelength  $\lambda = 70$  cm corresponding to the set FuG 101 a. Assuming the values  $f_m = 200$  c/sec,  $B = \pm 20$  Mc/sec  $= 40$  Mc/sec,  $r_{\max} = 50$  m and  $c = 3 \cdot 10^8$  m/sec then from equation (5), the frequency difference was

$$(6) \quad f_{D\max} = 5,300 \text{ c/sec.}$$

For use as an automatic fuse the "MARABU" was equipped with a simplified receiver having only two valves. It had a low frequency band pass width of about 300 to 5000 c/sec. Through the upper frequency limit a "mensuration" of the maximum response distance  $r_{\max} = 50$  m was feasible (proximity fuse).

The transmitter antenna consisted of a quarter-wave rod at the nose of the missile, which itself was excited as a "long-wire" aerial having a conical radiation pattern as shown in Fig. 8. Two shunted receiver antennae were

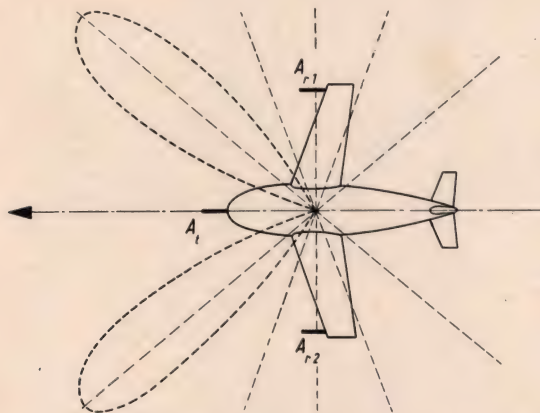


Fig. 8. "Marabu" radiation pattern

$A_t$  = Transmitter antenna

$A_r$  = Receiver antennae

located at the wingtips of the missile and together gave an axially symmetrical diagram. Therefore the system could be regarded as a type of directional fuse. As the important electrical components could be supplied from the quantity production of the FuG 101 a, the high apparatus expenditure was bearable.

In further projects it was intended to combine transmitter and receiver antennae as in the case of the "TRICHTER".

### 3.1.4. "Kugelblitz" (Patentverwertungs-Gesellschaft, Salzburg)

In the automatic fuse "KUGELBLITZ" of the PATENTVERWERTUNGS-GESELLSCHAFT, Salzburg, use was made of the fact that the base-resistance of an antenna can be varied by reflection from a reflector located in the neighbour-

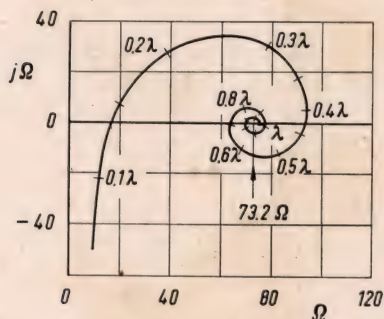


Fig. 9. Diagram of radiation resistance influenced by a moving reflector

$\lambda$  = Distance of reflector measured in wave length

hood, so that a valve-oscillator with unstable feed-back connected to the antenna-circuit is influenced. The locus-diagram in Fig. 9 shows e.g. the impedance of a half-wave dipole with reference to the current loop which is "interfered" by a half-wave reflector. In this case the feed-back effect can be quite easily determined by means of the values supplied by the antenna-theory for the radiation-coupling of shunted dipoles<sup>9</sup>. It emerges that, for very large distances (more than  $1.5\lambda$ ), the "normal" radiation resistance of  $73.2\Omega$  occurs, but that, as the reflector is approached, an increasing deviation from the normal value is produced in both the real and in the imaginary axes. The feed-back has active and reactive components. It is important that the reflector distance enters the unit  $\lambda$ , the feed-back increasing with increasing wavelength. On the other hand, a reduction of the wavelength  $\lambda$  will result in shortening the antenna and ameliorating the reflection-conditions<sup>4</sup>.

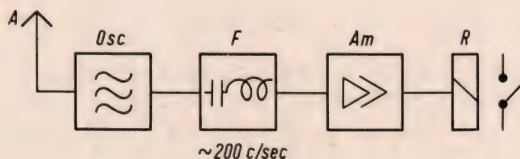


Fig. 10. Lay-out of automatic fuse "Kugelblitz"

A = Antenna                      Osc = Oscillator                      F = Filter  
Am = Amplifier                      R = Relay

The "KUGELBLITZ" (Fig. 10) worked with a transmitter in threepoint connection on the wavelength  $\lambda = 1$  m. The antenna, located at the nose of the missile, had a length of nearly  $\lambda/4$ . If the working point of the transmitter valve was well-chosen, then on approaching the target an anode-current variation in phase with the DOPPLER frequency was produced which was amplified in a

two-stage frequency amplifier and initiated the fuse. In all, three valves were required, one of which had to be a special VHF type. Contrary to the "MARABU" and "KAKADU" sets, this device had the advantage of requiring but one antenna. At first, it was doubtful whether, by producing in quantity and not using special valves, the sensitivity would be sufficiently uniform. Tests with a few thousand valves, however, showed the set to be serviceable when produced in quantity.

### 3.1.5. "Fox" (AEG, Berlin)

"Fox", the fuse constructed by the research laboratory of the AEG, Berlin (responsible engineer: Dipl.-Ing. K. HERZOG), (Fig. 11), worked exactly like the "KUGELBLITZ". However, it used a wavelength of  $\lambda = 3$  m. Whilst this fact

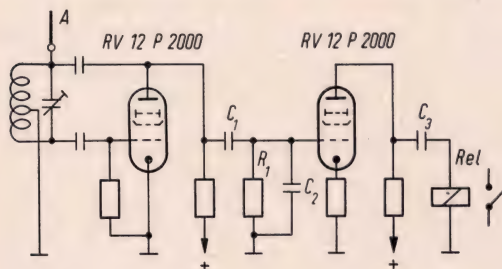


Fig. 11. Circuit of automatic fuse "Fox"

A = Antenna  
RV 12 P 2000 = Valves  
 $C_{1,2,3}$  = Condensers  
 $R_1$  = Resistance  
Rel = Relay

required an elongation of the antenna, the relative feed-back to the antenna was increased as can be gathered directly from the locus diagram in Fig. 9. Accordingly, only two valves were needed for the set so that on account of the lower operating frequency, a simple receiver valve could be used as input valve. Although the transmitter power was only a few tenths of one watt, the effective range was 12 to 15 m. The dimensions of the set were about 160 mm diameter and 100 mm long and it weighed — including the antenna — about 2.5 kg. The antenna had a length of 70 cm. Tests of the unit were, on the whole, completed, but it is unknown whether quantity production was started.

### 3.1.6. "Pinscher" (Ernst-Orlich-Institut, Danzig)

The high-frequency systems described below were developed by the ERNST-ORLICH-INSTITUT (EOI) of the "RHF" at Danzig <sup>4</sup>. All these systems operated on "long" waves in the range of about  $\lambda = 50$  m. Here, advantage was taken of the fact that particularly high reflection power could be obtained using the resonance wave of the target. The receiver input power is then:

$$(7) \quad N_r = 10^{-3} (\lambda/r)^4 N_t.$$

Numerous tests on the reflective properties of various aircraft and missiles had indicated that as a result of the small wave resistance of such spacious bodies, the damping of the radiation produced good broadband characteristics so that an accurate determination and maintenance of the resonance wave was not necessary. From equation (7) the input voltage  $U_i$  at the resonance resistance  $R$  of the receiver input circuit is

$$(8) \quad U_i = \sqrt{R \eta N_r} = (\lambda/r)^2 \sqrt{10^{-3} R \eta N_t}$$

In practice, the value of  $R$  is generally given by a diode-circuit with working resistance (e. g. telegraph-relay). The curves in Fig. 12 show that, with a transmitter output of one watt and with a system efficiency  $\eta$  of a few percent, even at a target distance of 30 m the reflected power gained according to equation (7) was large enough to produce, with a simple diode-circuit, working

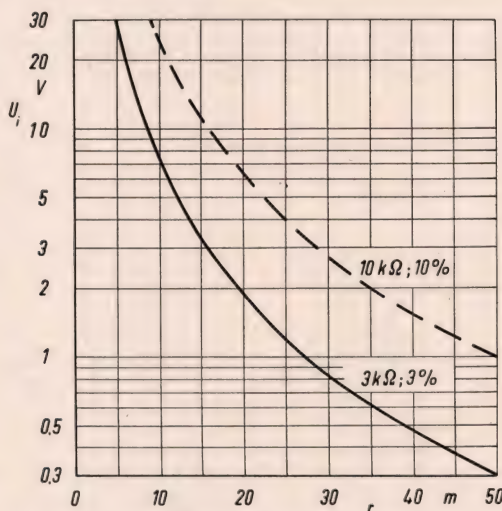


Fig. 12. Input voltage of automatic fuse "Pinscher"

$U_i$  = Input voltage

Transmitter output power  $N_t = 1\text{ W}$

Resonance resistance  $R = 3\text{ k}\Omega$  ( $10\text{ k}\Omega$ )

$r$  = Target distance

Wave length  $\lambda = 50\text{ m}$

Efficiency  $\eta = 3\%$  ( $10\%$ )

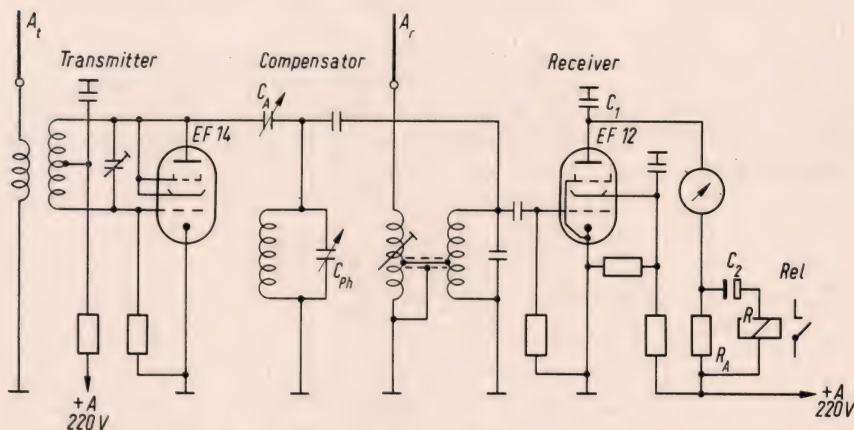


Fig. 13. Circuit of automatic fuse "Pinscher-B"

$A_{t,r}$  = Transmitter and receiver antennae

$R, R_A$  = Resistances

Rel = Relay

$C_{1,2,A,Ph}$  = Condensers

EF 12, 14 = Valves

voltages of the order of a few volts. Therefore a very simple transmitter — receiver system was sufficient. Even losses produced by aerials which were short in relation to  $\lambda$ , could be tolerated.

Fig. 13 shows e. g. the wiring diagram of the "PINSCHER-B" system consisting of a single-valve transmitter and a single-valve receiver. By means of a compensator (consisting of an LC circuit for phase adjustment of the compensating voltage), the coupling between the transmitting and the receiving antenna  $A_t$  and  $A_r$  could be reduced so that the receiver responded to the reflection from the target only. By means of a differentiation connection in the plate circuit (condenser  $C_2$  before the relay), the constant basic level by a direct pick-up from the transmitter (in the case of inaccurate compensation) was eliminated.

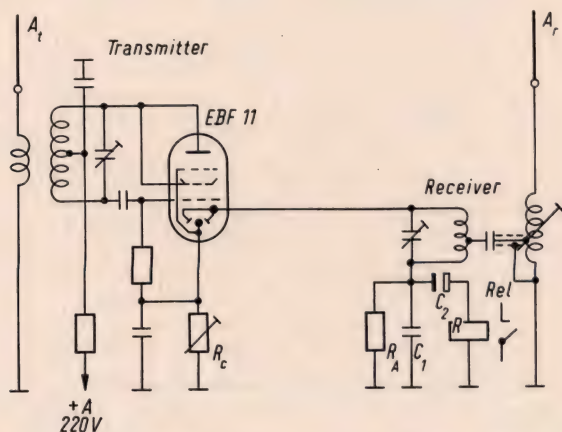


Fig. 14. Circuit of automatic fuse "Pinscher-E"

$A_{t,r}$  = Transmitter and receiver antennae      EBF 11 = Compound valve  
 $C_{1,2}$  = Condensers       $R, R_A, R_c$  = Resistances      Rel = Relay

A further example, without compensator, the "PINSCHER-E" shown in Fig. 14, operated with just one compound valve of type EBF 11. The receiving unit used the diodes of the transmitting valve and was de-sensitized to the residual coupling with the transmitter by a biasing potential (by means of the cathode resistance  $R_c$ ).



Fig. 15. Antennae mounting in the case of Hs 293 radio remote-controlled glider bomb

$A_{t,r}$  = Transmitter and receiver antennae

Ranges of 30 to 50 m could be attained with all these devices. It was a disadvantage of all "PINSCHER" types that two separate antennae were

necessary. Fig. 15 shows e. g. the projected mounting of the receiving antenna  $A_r$  for the HENSCHEL missile Hs 293 (consisting of a short rod at the nose of the missile) and the transmitting antenna  $A_t$  (consisting of a trailing wire several metres long at the tail of the missile). For distances up to 30 m oscillations  $\varphi$  of  $A_t$  amounting to  $\pm 10^\circ$  were considered as admissible.

### 3.1.7. "Wiesel" (Ernst-Orlich-Institut, Danzig)

The advantages of long resonance waves suggest using the direct reaction of the target on the transmitter for initiating the fuse, as illustrated in the examples of the "KUGELBLITZ" and "Fox".

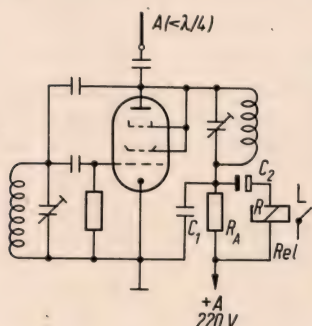


Fig. 16. Circuit of automatic fuse "Wiesel"

$A$  = Antenna  
 $C_{1,2}$  = Condensers  
 $R, R_A$  = Resistances  
 $Rel$  = Relay

Fig. 16 shows the wiring diagram of the fuse "WIESEL", developed by EOI at Danzig, using but one valve which worked on the principle of the so-called "HUTH-KÜHN Oscillator". This circuit is sensitive to the active and reactive components of the reflector feed-back. The fuse initiating was effected once again by a differentiation connection in the plate circuit of the valve. With an antenna only  $0.1 \lambda$  long, a range of 30 m was attained.

### 3.1.8. "Marder" (Ernst-Orlich-Institut, Danzig)

As has already been mentioned, the reflector feed-back (see the spiral curve in Fig. 9) contains a reactive component causing detuning of the antenna, provided the phase of the reflected radiation is suitable. In this way, a frequency change of the oscillator feeding the antenna can be achieved. This allows the connection of two oscillators  $G_1$  and  $G_2$  as a "heterodyne oscillator" (Fig. 17). A variation in the beat frequency (by detuning of  $G_1$ ) initiates the fuse.

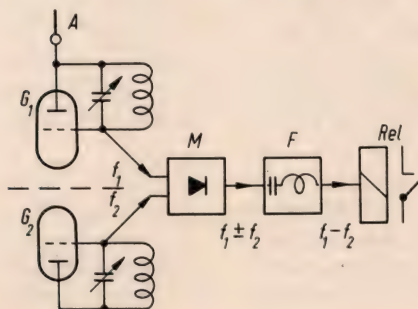


Fig. 17. Principle of fuse-initiating by frequency change caused by reflected radiation

$G_{1,2}$  = Oscillators  
 $f_{1,2}$  = Frequencies  
 $M$  = Mixer  
 $F$  = Filter  
 $Rel$  = Relay

In the "MARDER" system (EOI Danzig), the well-known difficulties in operating a heterodyne oscillator (oscillator inconstancy, mutual carrying) were avoided by the arrangement shown in Fig. 18. The unit fitted into the missile consisted only of a self-exciting single-valve oscillator of the simplest construction with a short rod or trailing wire antenna. The feed-back from the target produced a small frequency deviation which could be transformed by

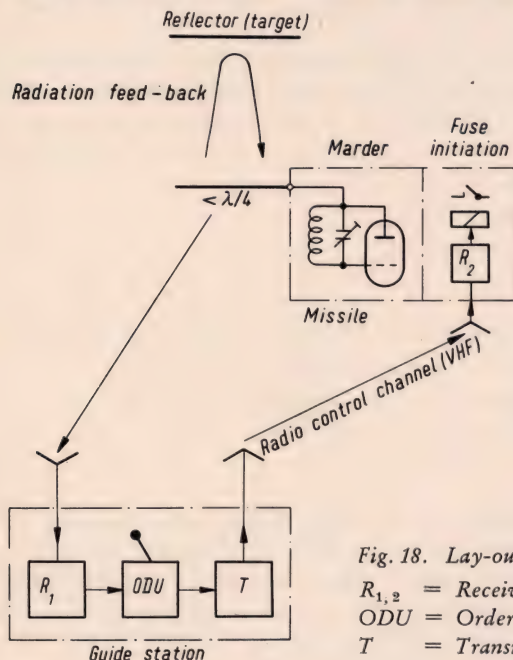


Fig. 18. Lay-out of automatic fuse "Marder"

$R_{1,2}$  = Receivers

ODU = Order distributor unit

$T$  = Transmitter

the control receiver at the guide station into an automatic initiation of the fuse ignition. To accomplish this, an initiation command was given to the order distributor unit ODU from the "MARDER" receiver  $R_1$ , and transmitted on the radio control channel via the transmitter  $T$  to the missile. A separate radio-channel with a fifth control frequency was already envisaged for the normal radio-control sets as e.g. "KEHL/STRASSBURG"<sup>1</sup>. Slow frequency deviations in the "MARDER", caused e.g. by thermal effects or by variations in the operating voltage, were counteracted by automatic sharp tuning of the control receiver. This tuning, however, worked with a time lag, so that rapid frequency deviations on passing the target were fully effective. The advantage of this system consisted in the extremely straightforward construction of the part which was lost in action, and which operated with any valves and did not require balancing. With such a system, using an EF 14 valve, ranges of about 30 m and more could be attained.

Some additional fundamental comments on the "long wave" systems ( $\lambda \approx 50$  m) may be in order. The reflector, excited by resonance, emits, like a half-wave dipole, an approximate cosine diagram. As is known, this diagram enters quadratically, i.e. on reception and on reflection. Together with the cosine diagrams of the missile antennae a cosine function of a fourth power,

i. e. a very suitable directional fuse, is obtained. This was fully proved in flight tests. Furthermore, it was found that metal foil did not disturb the fuses even when the missile passed through just-spread concentrated metal-foil clouds <sup>4</sup>.

### 3.2. Electrostatic Fuses

*"Kuhglocke" (Rheinmetall-Borsig, Breslau)*

As an example of an electrostatic proximity fuse the "KUHGLOCKE" system, developed by RHEINMETALL-BORSIG, Breslau (designed by Dr. RIEDEL, following a suggestion by Prof. VIEWEG, Technical University Darmstadt), may be mentioned. It made use of the fact that an aeroplane, due to the ionized exhaust gases of the engines, acquires an electrostatic charge of many thousand volts which gives rise to a corresponding field-intensity around the aircraft.

A small low voltage neon glow lamp, charged just up to below the discharge point by some voltage source, was attached to the nose of the missile, as shown in Fig. 19. The glow lamp was switched between the body of the missile and

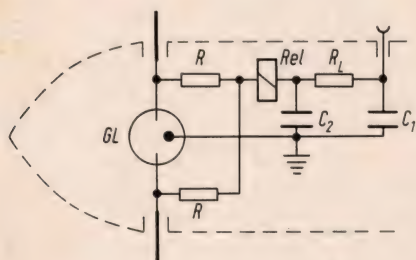


Fig. 19. Lay-out of automatic fuse "Kuhglocke"

$R, R_L$  = Resistances

$C_{1,2}$  = Condensers

GL = Low voltage neon glow lamp

Rel = Relay

one or more insulated electrodes which protruded from the missile. If an electric charge was located laterally from the rotating missile, an alternating voltage, the frequency of which corresponded to the rotating frequency of the missile, was originated at each electrode of the glow-lamp. This alternating voltage initiated the lamp, thereby closing the relay circuit and initiating the fuse. Since no current was required prior to the initiation, the operating voltage could be supplied simply by a condenser  $C_2$ , which was charged on firing (by means of the safety circuit consisting of  $C_1$  and  $R_L$ ). This simple device could be fitted into the heads of missiles of 10.5 cm calibre and more. The obtainable detonation range was about 3 to 5 m. A further increase of sensitivity frequently led to unexpected detonations due to the electrical inhomogeneity of the atmosphere.

Therefore, this method is limited and is feasible only for smaller missiles. In addition to the increase of the area of impact, the advantage of such a fuse was to be found in the initiation by impact on very light targets, where a normal percussion fuse, due to insufficient retardation, might not have functioned.

### 3.3. Magnetic Fuses

The magnetic fuses made use of the fact that the homogeneous magnetic field of the earth is inhomogeneous (i. e. it will become distorted) in the neighbourhood of a ferromagnetic body. This effect emanates e. g. from the iron parts of an aircraft engine. The change of the earthfield will lead e. g. in

an induction coil to induced electromotive forces which may be used for initiating the fuse. The difficulty connected with this procedure, however, is due to the fact that induced electromotive forces will also be obtained when the direction of the axis of the induction coil is changed within the magnetic field, e. g. by a change of the flight-path of the missile. Thus it becomes imperative to provide a system of two astatic coils where induced electromotive forces caused by a change of direction will eliminate each other.

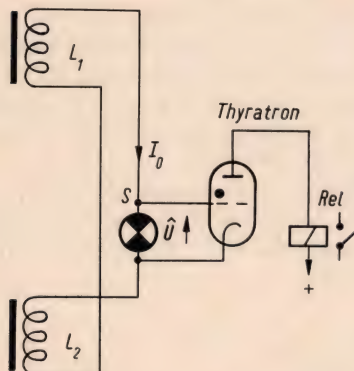


Fig. 20. Lay-out of automatic fuse "Isegrimm"

$L_{1,2}$  = Induction coils  
 $S$  = Switch  
 $I_0$  = Induced current  
 $\hat{U}$  = Voltage pulse  
 $Rel$  = Relay

The magnetic fuse "ISEGRIMM" (ERNST-ORLICH-INSTITUT, Danzig) (Fig. 20) showed an interesting working principle: it had two induction coils  $L_1$  and  $L_2$ , the short circuit of which was periodically interrupted by the switch  $S$ . When passing a magnetically effective target, there occurred as result of the induced electromotive force  $E$  a short-circuit voltage  $I_0 = E/R$ , and on interruption of the current there was obtained, owing to the effect of the self-induction  $L = L_1 + L_2$  (with the accumulated energy  $\frac{1}{2} L I_0^2$ ), at the switch-terminals a voltage pulse

$$(9) \quad \hat{U} = I_0 Z = I_0 \sqrt{L/C}.$$

Thus, there resulted<sup>12</sup> an "amplification"  $A = Z/R$ . The voltage  $\hat{U}$  could be used to ignite a Thyatron, which in its turn initiated the fuse relay  $Rel$ .

All developments regarding magnetic fuses, however, were stopped, because a satisfactory suppression of the interference level was giving too much trouble.

### 3.4. Acoustical Fuses

All acoustical fuses presented the problem of separating the effective noise (engine noise of the flying target) which was to initiate the fuse, from the interfering noise of the slip-stream and of the propulsion unit of the missile itself. Numerous investigations showed that the distance between effective noise and interfering noise, mainly in the range between 100 and 200 c/sec, could be made sufficiently large.

#### 3.4.1. "Meise" (Neumann & Borm, Berlin)

Thus, a response distance of nearly 15 m was attained in the "MEISE" system, developed by NEUMANN & BORM Company, Berlin. It consisted of a resonance

microphone and a two-stage amplifier and was intended for use against propeller-driven aircraft.

### 3.4.2. "Kranich" (Ruhrstahl A.G., Brackwede)

The "KRANICH" fuse built by RUHRSTAHL A.G., Brackwede (responsible engineer: Dr. M. KRAMER), was particularly noteworthy owing to its simple design. It consisted of a membrane with an inherent frequency of 200 c/sec built into the nose of the rocket. A steel wire fixed in the middle of the membrane and tuned by proper dimensioning, also to 200 c/sec, was set into vibration by the noise emitted from the target. A contact ring fitted around the tip of the wire closed a contact even in case of small vibration amplitudes and initiated the fuse. The unit had a response range of 7 m and, of course, this range was considered adequate for small rockets only.

### 3.5. Optical Devices

#### "Paplitz" (Electroacoustic Comp., Kiel)

As an example of the optical developments mention should be made of the "PAPLITZ", an infra-red fuse designed by the ELECTROACOUSTIC COMP. at Kiel (responsible engineers: Dr. E. W. KUTSCHER and Dipl.-Ing. ORLICH). The unit was supposed to respond to the heat radiation of the hot exhaust gases of aircraft<sup>3</sup>.

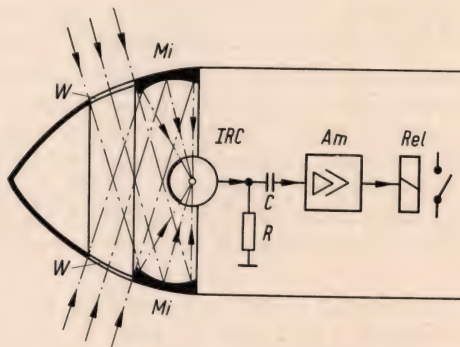


Fig. 21. Lay-out of automatic fuse "Paplitz"

W = Windows of quartz-glass	Mi = Mirror	IRC = Infrared cell
R = Resistance	C = Condenser	Am = Amplifier
Rel = Relay		

Fig. 21 illustrates schematically its operation. Just below the tip of the missile there was an annular window W made of quartz-glass. Inside, behind the window, there was an annular mirror Mi, focussed exactly on to an infrared cell IRC located on the missile axis. The system, therefore, was to be regarded as a directional fuse with forward-inclined conical radiation characteristics, corresponding to the "direction of view" of the infra-red cell. A two-valve amplifier was connected in series with the working resistance R of the cell behind a high-pass filter. The amplifier system was so dimensioned, with regard to the frequency of its bandwidth, that an undesired initiation by permanent

light could be prevented. The response range of the system was about 25 m. The device worked quite well when tested, but it could be used only at night, because variations of the interference radiation by daylight due e.g. to sharply limited clouds made it impossible to reliably prevent unexpected detonations.

#### 4. CONCLUSIONS

A critical comparison of the systems described above leads to the following conclusions: there are numerous physically different possibilities for automatic fuses, which could be technically realized. In future, this fact alone will not be a sufficient criterion, and more than ever before the problem will arise as to how intentional jamming might be overcome.

Undoubtedly, the high-frequency systems are particularly elegant and reliable with respect to "natural" interference. Unfortunately, the long-wave systems are particularly liable to jamming by interference transmitters and, in addition, the short-wave systems are susceptible to jamming by "Chaff" (metal foil). Jamming of the units might be prevented by introducing a very selective transverse directivity. This method would demand a change-over to the shortest wave-lengths which today can already be controlled by modern techniques. In addition, a qualification of the target area should be introduced to make "chaff" ineffective.

Experience with electrostatic and magnetic fuses has shown that, due to the interference level, only small working ranges can be realized. Such fuses, therefore, are out of consideration for large guided or homing missiles.

Acoustical fuses cannot actually be interfered with intentionally, since it is impossible to produce sufficiently high interference noise levels. These units can easily be made insensitive to the effect of sudden high peak intensities (e. g. detonation of a shell). However, all acoustic systems are deficient in as far as the speed of sound can no longer be neglected in relation to the cruising speed of the flying target (the speed of the missile is not important in passive systems). In future, this fact might become decisive, especially when fighting high-speed rockets.

Due to natural interference, the infra-red technique has also led to some difficulties. Today it might be possible to built active infra-red systems operating on very long waves ( $\lambda > 15 \mu$ ), where interference influences might be cut out by "coding" (e. g. by modulation). Therefore, the border region between infra-red and high-frequency technique should demand particular interest.

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# SOME SPECIAL PROBLEMS OF POWER PLANTS

OTTO LUTZ \*

## 1. PROPELLANT PROBLEMS

In the early days of rocket propulsion, the right choice of the propellant was of vital importance for quick and successful development. In a doctor's thesis, my former collaborator Dr. NOEGGERATH has compiled all practically applicable reactions starting from the thermodynamics side. What is generally known today would have been a certain surprise to us 20 years ago, namely, that there are no combinations of chemical propellants which are exceptionally better in output of energy than others. As for the highly desirable low molecular weight, there are no outstanding ones among the practically applicable combinations of propellants.

In 1935 we had already dealt with the question whether a simplification of rocket design could be achieved by co-operative research in matching the point of view of the chemist with the demands of the design engineer. We searched to find processes and propellants which would reduce the engineering difficulties. Or, in other words, the effort required to gain the maximum energy output was not to be devoted to engineering only, but also, by the use of special mixtures, to the process or the propellant.

This was to be the guiding principle for power plants designed for use in great numbers. Later, the idea proved to be of importance for the entire field of rocketry.

### 1.1. Monergoles (Mixtures of Propellants)

I have to refer to some thoughts which had already been published in 1943, but which did not become known generally. Although it is not of great importance, I should like to mention here that all propellant names ending with "ergol" were created by us at Brunswick. Dr. NOEGGERATH developed an extraordinary ability for such telling nomenclature. Every gunpowder is, in a sense, a monergolous mixture of propellants. This paper will deal only with liquid monergoles, i. e. with such mixtures of propellants in which the oxidizer, the fuel, and the ballast materials are contained — forming a homogeneous mixture. Difficulties arise on the one hand because the propellant combination has to react easily and completely in the reaction chamber, and on the other hand because the flash back of the reaction into the storage tanks has to be avoided under all circumstances.

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One of the ways of finding an answer to these contradictory demands is given by the application of substances which possess high excitation energies in the wider meaning of the word, e. g. heat of evaporation, positive heat of formation. Our first trial to use ammonia as the fuel and ammonium nitrate as the oxidizer, the so-called "Divers' Liquid", could easily be governed from the point of view of safety, yet its corrosiveness and the fact that the mixture tends to separate brought up new difficulties. By replacing the ammonium nitrate with nitrous oxide one can get over these difficulties. The engine could be made explosion-proof by installing high heat absorbing material in the piping system, but no final security against shock waves caused by detonations could



Fig. 1.

*Interruption of a monergol explosion in a duct by built-in explosion safety device*

be achieved (Fig. 1). These reasons compelled us to abandon experiments although we believe that due to the extraordinary simplicity of this type of engine the monergoles will always remain important for certain special cases of application.

## 1.2. Lithergoles

The next step was to leave one component as a solid in the reaction chamber and to feed in the other one as a liquid, gas or vapour. A suitable arrangement of the reacting surfaces (controlling by interchange distances and time, or by increased diffusion by using forced turbulence) allows the fluid component to pick up just as much of the fixed solid component as is necessary to maintain a satisfactory continuous reaction. This process also results in extremely simple engine design. It was ANDRUSSOW who proposed putting the fuel in the reaction chamber in the form of coal and feeding in the oxidizer in the

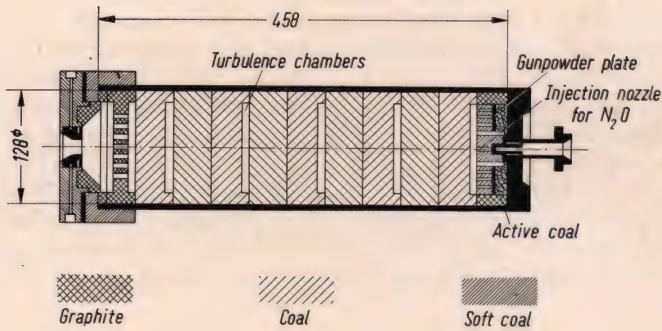


Fig. 2. Lithergol-combustion chamber

Maximum thrust is attained in 1.2 sec

form of nitrous oxide. We shall deal again with nitrous oxide later. Fig. 2 is a cross section through an experimental engine showing the coal charge consisting of single discs with holes drilled in an axial direction. The outer shell does not need heat protection, as coal has a very low thermal conductivity. The only precaution one had to take was not to locate holes too near to the shell; that meant leaving the area nearest to the circumference unperforated. Ignition was achieved in a most simple manner by a small charge of gunpowder which ignited a quantity of granulated active coal. Intensive study had to be devoted to get the reaction taking place simultaneously over the full length of the intake, thus getting the charge burning off not from one end to the other but in a radial direction. That was obtained by a lining of celluloid inside each hole, which heated the entire inside surface to ignition temperature instantly. Full



Fig. 3. Simplified produced coal charges

thrust could be gained within one second and thrust oscillations could be reduced to less than 5 per cent. The coal charge shown in Fig. 3 is a bundle of small coal tubes sintered together in order to simplify production. The even combustion is shown in Fig. 4.

Such simple engines are suitable for thrusts of about 500 to 1000 kg and 40 seconds' to 2 minutes' burning time. The length of the engine is determined by the period of the optimum fuel flow velocity, which means that a further increase of thrust can be achieved only by multiple assemblies in



Fig. 4. Coal charge of a lithergol-engine, partly burnt out

parallel, or, by what is equivalent, an enlarged diameter. It is quite understandable that this method sets a limit to the overall impulse of such engines which, however, can be favourably applied in such cases where the combustion periods exclude powder rockets, yet the specific kind of use requires an engine almost as simple as the latter.

### 1.3. Hypergoles

#### 1.3.1. History

Whilst in the cases described above we tried to simplify the entire power plant by the selection of an especially favourable process, with hypergolic processes we concentrated the necessary effort on the propellant.

We started here with the following thoughts:

A very intimate mixing of fuel and oxidizer has to be aimed at in order to keep the necessary reaction space as small as possible. As soon as a reaction stagnates, however, mixtures of fuel and oxidizer are formed, which in most cases must be considered as highly explosive. If, in addition, a reaction ceases due to insufficient heat supply, extremely explosive by-products — as for instance peroxides — can be formed. Considering the extraordinarily large rates involved, it is easily understood that even during the shortest delay very considerable amounts of explosives can be formed in the reaction chamber. These facts demand, especially in the case of controlled power plants, quite expensive design features. If, however, hypergolic propellants are used, i. e. substances which react without any energy being added spontaneously and completely as soon as they come into contact — the above mentioned danger is avoided without any additional expense, provided ignition delay is short enough. This is of special importance for power plants used at high altitudes.

It might not have been of great merit to have proposed this idea, but I had the good fortune to have colleagues who — after I had mentioned this principle in 1935 — worked for years and years with intuition and never

failing energy towards realization of this idea. I have to mention here Dr. HAUSMANN, Dr. NOEGGERATH and Dipl.-Ing. EGELHAAF.

### 1.3.2. *General Aspects of Hypergoly*

Hypergoles — this name too was given by NOEGGERATH — must not be understood as combinations of substances which decompose by releasing their negative energy of formation only, but as combinations of at least two substances which react directly with each other forming gaseous or vaporous reaction products.

The application of hypergolyous propellants has, apart from the above mentioned advantages, also one disadvantage. If for any reason, as for instance tank leakage caused by defect or enemy action, the components come accidentally into contact with another, they will ignite. The propellants, however, will only ignite and burn on their contacting surfaces and will not form any large amounts of explosives, which would be the inevitable result with non-hypergolyous propellants.

Experience has shown that the application of hypergolyous propellants did not increase but reduced the safety risks. The world's first manned aircraft surpassing a speed of 1000 km/h was powered by our hypergole "hydrogen peroxide-hydrazine hydrate".

The requirements for hypergoles are numerous, and one of our main tasks was to find out precisely, and to compile, all these requirements, and to define, if possible, characteristic coefficients which could express to what extent the requirements could be met.

The chemical properties of hypergolyous substances have to be considered first. And here we meet the first contradiction; on the one hand the substances involved have to show an extreme affinity, whereas on the other hand they should be storable for unlimited time, even under adverse conditions. This makes it clear how difficult the problem is and that it will not be easy to determine, by means of basic theoretical considerations only, whether or not both demands can be met.

I only draw attention here to: dehydrations, redox-reactions, addition reactions, creation of complexes, steric factors, polymerisations, condensations, forming of salts.

To most of the hypergolyous propellant combinations a catalytic agent has to be added in order to shorten ignition delay; in all cases tested so far, use was made of a metallic catalyst dissolved in one of the components. The question of solubility and stability of the catalyst was in itself a field of study.

Mention can be made here only of the most important of the necessary physical properties such as the solidification and decomposition points, the viscosity, especially at low temperatures, and the vapour pressure which might be of importance (filling, vapour locks in pumps or ducts and so on). Furthermore, the surface tension, the characteristic coefficients for heat transfer, etc. are of interest. The greatest difficulty arose from the solidification and decomposition of these substances because in most cases we had to deal with mixtures of a number of components.

The thermodynamical properties of the substances do not generally show too great differences. As already pointed out, the energy content has to be

considered from different angles, depending upon the admissible temperature inside the reaction chamber. The significance of the density of the mixture has already been mentioned. Apart from these purely material properties other points such as price, availability, physiological influences, can be decisive. The availability of supplies becomes naturally decisive for application if all other properties fulfil to a sufficient extent the defined demands.

It is to HAUSSMANN's merit that he drew attention to the hypergolic combination, hydrogen peroxide and hydrazine hydrate, as early as 1935. In the case of nitric acid we used also amino compounds first, as they did satisfy to a high degree the specific properties of the oxidizer, and they seemed to be very suitable for the excitation of the reaction. Besides these we later applied a large number of other substances. If hydrogen peroxide is used — which itself does not contain nitrogen — nitrogen-free substances become particularly interesting for naval use, as they produce reaction gases which leave no tracks in water. Fuels which consist mainly of vinyl-ethers have proved very suitable for reaction with nitric acid. These substances, which we called "visoles", could be used favourably with mixed acid as oxidizer, due to its better corrosion properties than pure nitric acid.

Almost all the proposed hypergolic propellants consisted of mixtures of different compounds. This results, of course, in a complication of the individual effects, yet mixing offers just the possibility to intensify one or the other desired property. For instance, the chemical affinity of a mixture of two substances is in some way analogous to the solidification diagram of a system. Fig. 5 shows this affinity expressed as a limit concentration, i. e. the acid concentration at which ignition takes place without perceptible delay.

It can easily be recognized that mixtures may have a considerably higher affinity than the single components, an effect which has also been proved true with numerous other substances. The same diagram shows the lowest admissible

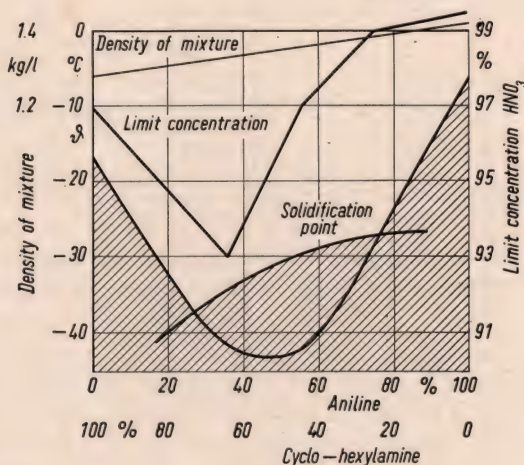


Fig. 5. Characteristic values of the hypergolic fuel system aniline and cyclo-hexylamine  
Limit concentration = acid concentration up to which no delay is noticed

temperature, the so-called "cold point". This "cold point" is given at both ends of the diagram by the solidification point, in the middle by the highest admissible viscosity, which was assumed to be 40 centi-stokes for a particular case. In this special case the optimum in regard to "cold point" as well as to ignition delay are almost identical. There are, however, combinations of substances showing the contrary; we then have to try to bring both optima into accordance by adding further components.

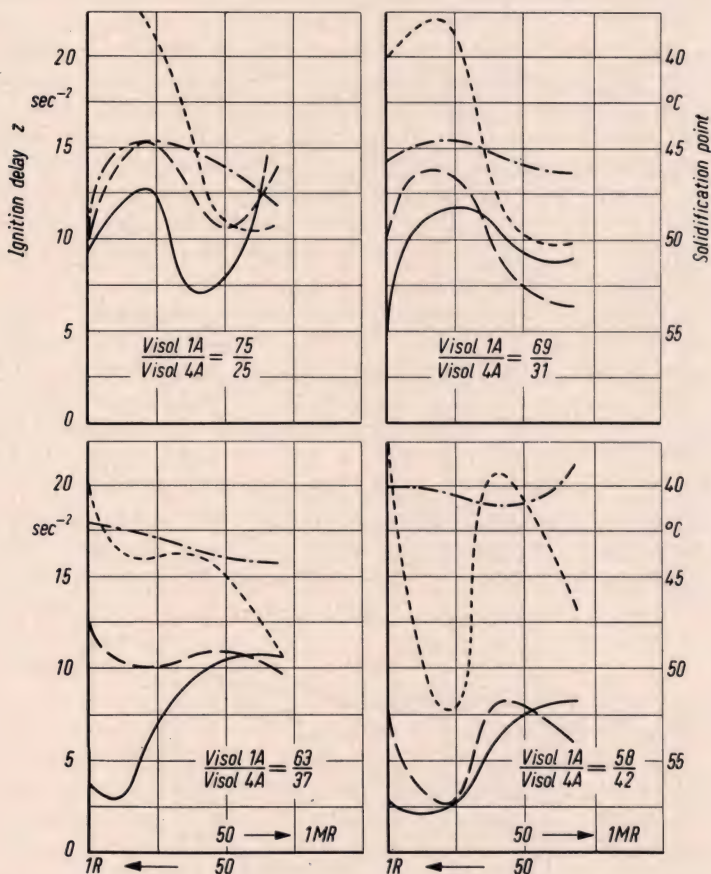


Fig. 6. Influence of the composition of the visol and amine component in visol fuels  
Proportionality factor = parts by weight

Fig. 6 shows different diagrams obtained in the development of "visol" fuels. We are dealing here with a mixture of 4 components: 2 different visoles, the vinyl-butyl-ether (visole 1) and the butane-diol-divinyl-ether (visole 4) and 2 different organic amino compounds, aniline and methyl-aniline. The ignition delay is shown as a function of the composition of the amino mixture. The dotted lines correspond to the substances with 10 parts by volume, the broken lines to the substances with 15 parts by volume and the solid lines to the

substance with 20 parts by volume of amino mixture. Finally, the four diagrams differ in their visole composition. Without going into more detail, as for instance the conversion of a maximum into a minimum by changing the composition of the visole part, I want to draw attention to the extraordinary differences caused by small changes in the absolute contents of the amino components. This sensitivity made studies very difficult from the point of view of affinity; the systems show all properties of multi-component systems and the technician is tempted to speak of a eutectic.

It should also be mentioned that the ignition delays given do not claim to be absolute values. Ignition delay is more or less influenced by the way in which the components are brought together. We might even arrive at points where an arrangement which is favourable for one combination of substances results in long ignition delays for another fuel system.

The ignition delay times were measured by a photo electric cell by feeding the oxidizer uniformly and reducibly into the fuel, which was kept in a crucible.

We studied about 1100 hypergoles, but it would carry us too far to give more than a few of them here. To make a long story short I should like to sum up as follows:

### 1.3.3. *Hypergoles for Hydrogen Peroxide, the Different Developments, Hydrazine Hydrate, Substitutes, Results*

- a) For low percentage hydrogen peroxide:  
50%  $N_2H_4 \cdot H_2O$ , 47% methanol, 3% water plus 0.3% colloidal copper.
- b) For high percentage hydrogen peroxide:  
30%  $N_2H_4 \cdot H_2O$ , 57% methanol (called "C-stoff"), 13% water and traces of cupro-potassium-cyanide or colourless dissolved copper oxide.
- c) Other hypergoles for  $H_2O_2$ :

Hydrazine hydrate substituted by aliphatic amino compounds: Diethylene-triamine, ethylene-diamine, triethylene-tetramine with a copper sulphate catalyst, show good ignition properties, and the most important one of their physical properties is a high viscosity. However, their behaviour in the cold was found to be unfavourable.

Aldehydes (with vanadium or iron as catalyst), also show good ignition properties but are not as good as hydrazine hydrate.

Liquids normally used as developers, such as hydroquinone and pyrocatechol in a methanolic solution and with iron as catalyst, gained importance as "Optol-Fuels" which were available in greater amounts as chemical by-products and were taken into consideration to ensure a broader fuel basis for the Me 163 fighter plane. There were good results (EGELHAAF).

In summing up it can be said that "T-stoffs" could not, even after intensive study, yield results as good as those obtained with hydrazine hydrate.

### 1.3.4. *Hypergoles for Nitric Acid, Amino Compounds, Unsaturated Compounds, Developers, Other Developments*

- a) Aliphatic amino compounds: E. g. diethylene-triamine, poly-alkyl-poly-amines, triethylamine, methylamine; these reacted very well with ordinary nitric acid, as well as with nitric acid to which iron or vanadium catalysts were added, or with mixed acid (MS 10).

- b) Aromatic amino compounds: Starting with cyclo-hexylamine, the following amines proved to be especially suitable: aniline and its mixtures with other aliphatic or aromatic amino compounds (triethylene-amine, cyclo-hexylamine, methylaniline, pyridine, ethylaniline, xylidine, piperidine, pyrrole).

Certain mixtures show a "eutectic hypergolyty".

The hypergolyty of the above compounds is so good that dilutions with inert fuels have been possible.

The group of hypergoles mentioned was called "Tonka" by BMW and at Brunswick, "Gola". The BMW research staff conducted studies themselves in this field of hypergoles with excellent results (Fig. 7).

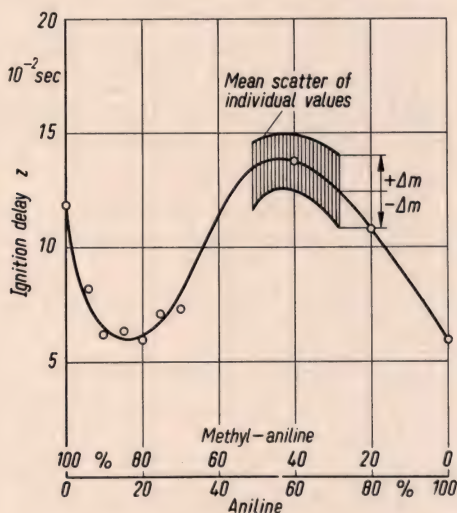


Fig. 7. Ignition delays of the hypergolic fuel "Gola"  
Mean values of 10 individual values

- c) Unsaturated compounds: Substances belonging to the acetylene group (Dr. REPPE) as, for instance, di-acetylene.

Vinyl-ethers: vinyl-ethyl-ether, vinyl-iso-butyl-ether, butane-diol-divinyl-ether, divinyl-acetylene, diketenes, cyclo-pentadine.

The hypergoles of the vinyl-ether group were called "Visoles" and were mostly used in combination with amino compounds.

- d) Developers: Pyrocatechol, hydroquinone, pyrogallol, and, in addition, "optoles". The components suitable for hydrogen peroxide proved to be suitable also for nitric acid.

- e) Others: Furan and derivatives, in particular furfuryl-alcohol, called "Fantol" (EGELHAAF).

They have particularly good hypergolyty, especially with mixed acid, even when diluted to a high degree with up to 70% xylol. Hydrazine also behaves hypergolytically with nitric acid.

## 1.3.5. Remarks on the Ignition of Hypergoles, Design of Mixing Injectors

It is typical that generally the hypergoles cannot be ignited by quick and intimate mixing in the stoichiometric ratio. If, however, the components are brought together in such a way that mixing can take place only at the relative boundary surfaces, the hypergoles will ignite spontaneously.

Whether a certain mixture ratio of the components, which might be very different from the stoichiometric combustion ratio, is necessary for ignition, could not be finally solved (EGELHAAF, studies not completed).

From the development of suitable mixing injectors resulted the principle that the energy used for atomizing has to be kept low. In general, mixing arrangements which brought together both flows with a small amount of energy but with split-up boundary surfaces proved to be good (EGELHAAF).

Formulae:

Di-acetylene:  $\text{HC} \equiv \text{C} - \text{C} \equiv \text{CH}$

Vinyl-ethyl-ether:  $\text{H}_2\text{C} = \text{CH} - \text{O} - \text{C}_2\text{H}_5$

Vinyl-iso-butyl-ether:  $\text{H}_2\text{C} = \text{CH} - \text{O} - \text{CH}_2 - \text{CH}(\text{CH}_3)_2$

Butane-diol-divinyl-ether:

$\text{CH}_2 = \text{CH} - \text{O} - \text{CH}_2 - \text{CH}_2 - \text{CH}_2 - \text{CH}_2 - \text{O} - \text{CH} = \text{CH}_2$

Divinyl-acetylene:  $\text{H}_2\text{C} = \text{CH} - \text{C} \equiv \text{C} - \text{CH} = \text{CH}_2$

Diketene:  $\text{C} = \text{C} = \text{C} = \text{C} = \text{O}$

## 2. CHEMICAL MEANS OF INCREASING THE OUTPUT OF CYCLIC POWER PLANTS

We also applied nitrous oxide to increase the high altitude output of piston engines. As our work revealed a number of interesting problems which are perhaps of some importance to other applications — in particular to inter-

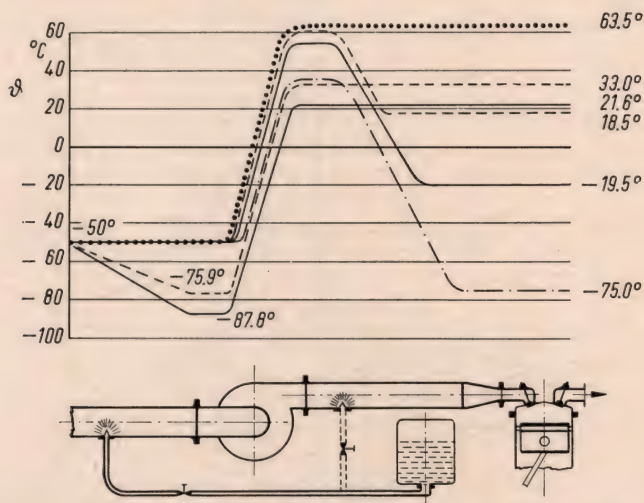


Fig. 8. Temperature distribution in the air intake to the supercharger

..... Normal operation    ——— With  $\text{N}_2\text{O}$     - - - With  $\text{O}_2$     - · - With  $\text{H}_2\text{O}_2$

mittently operating jet engines (pulse jets) — I should like to make a few comments.

Fig. 8 shows such an arrangement schematically. Liquid nitrous oxide is injected into the air intake of the supercharger, thus considerably cooling the

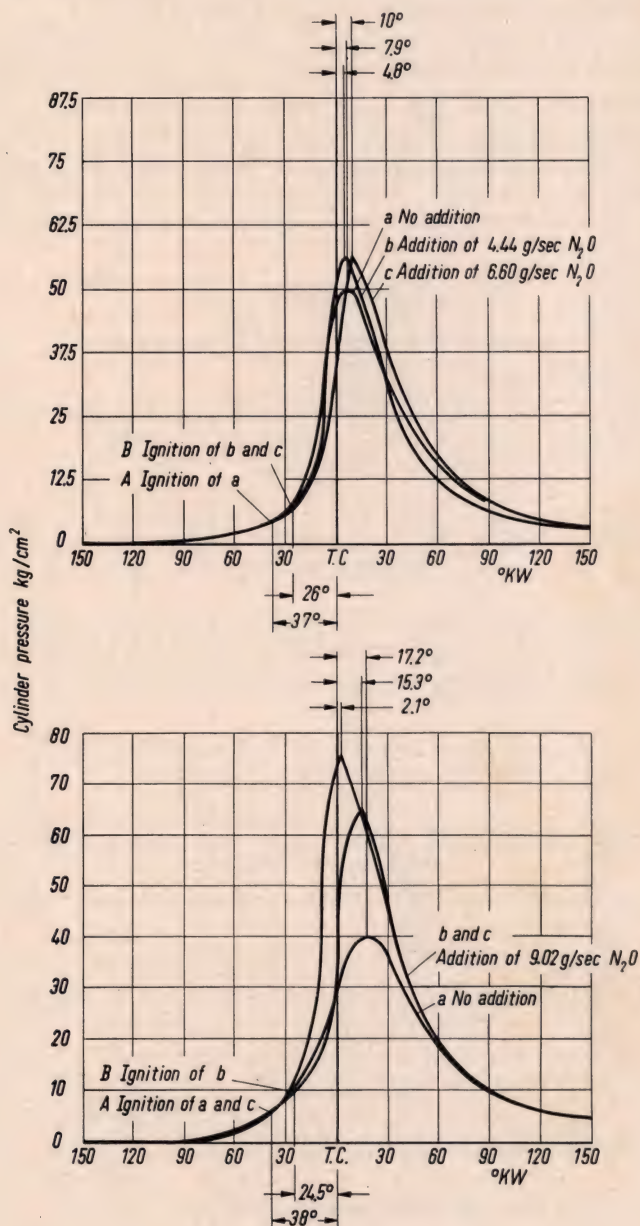


Fig. 9. Pressure-time diagrams  
T.C. = Top centre °KW = Crankshaft angle

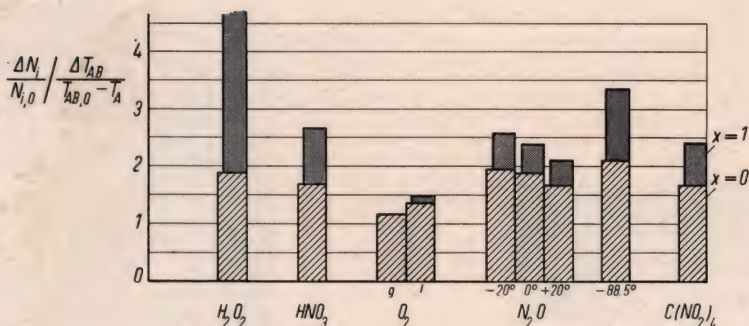


Fig. 10. Ratio of increase in output to increase in exhaust temperature

$g$  = gaseous

$l$  = liquid

$x = 0$ : liquid

$x = 1$ : completely vaporized

air to be compressed, with the immediate result that the supercharger delivers a greater mass of air to the motor. The cylinders obtain an air-fuel charge weighing more at a lower temperature. The increase of energy output is due in part to the heat of decomposition of the nitrous oxide, and in part to the combustion of the additional quantity of oxygen. The thermodynamic and physical properties of nitrous oxide suit the requirements of a piston engine in such a fortunate manner that the high altitude output of the aero-engines of that time could be increased by about 100% without any need for additional accessories for cooling or fuel injection. Variation of ignition timing (Fig. 9) enabled the maximum pressure to be kept practically constant. Compared with other oxygen carriers nitrous oxide offers the most favourable output increases in relation to the thermal load (Fig. 10). The specific output yields 4 PS per g/sec of nitrous oxide (Fig. 11).

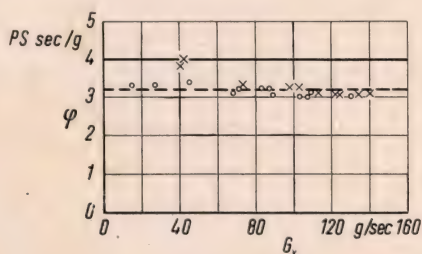


Fig. 11. Specific output

- Theoretical values for liquid undercooled stuff
- - - Theoretical values for stuff liquefied by pressure
- × Test results for liquid undercooled stuff
- Test results for stuff liquefied by pressure

For the application in fighter planes the nitrous oxide was carried aboard in high pressure cylinders and fed into the engine by its own vapour pressure (Fig. 12). One problem arose from the fact that the cylinders were not bullet-proof. I was able to find a very simple solution (Fig. 13) by placing a special

inner container (non-pressurized) in the middle of the bottle: the liquid was thus surrounded by a vapour chamber, which served as an elastic buffer, and the bottle became absolutely bullet-proof. Twin-engined reconnaissance planes carried the nitrous oxide aboard as a non-pressurized liquid at about  $-90^{\circ}\text{C}$  (Fig. 14).



Fig. 12.  $\text{N}_2\text{O}$ -booster unit built for Me 109 F

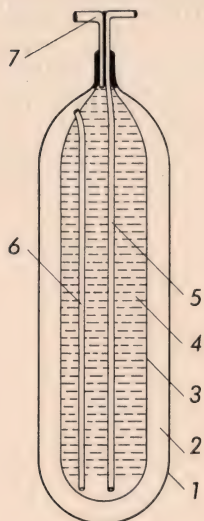


Fig. 13. Liquefied gas container with parts built-in

- |                                   |                          |
|-----------------------------------|--------------------------|
| 1 = Usual liquefied gas container | 2 = Vapour chamber       |
| 3 = Inner container               | 4 = Liquid chamber       |
| 5 = Suction passage               | 6 = Compensating passage |
| 7 = Outlet passage                |                          |

With the help of this chemical process for output increase, twin-engined bombers gained at an altitude of about 8 to 9 km such an increase in speed that they became faster than any British fighter for a time of 45 minutes, which was tactically sufficient.

All special reconnaissance and high altitude missions from 1943 to 1945 were flown with nitrous oxide booster units.

We also applied the nitrous oxide to the V-1 pulse jets; the thrust here could be increased by about 30%. The practical application did, however, not reach operational use.

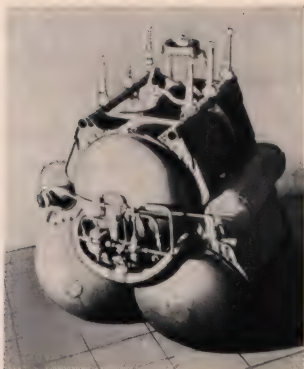


Fig. 14. Booster unit for increase in output

### 3. SHORT REMARKS ON SPECIAL MATERIALS, COMPOUND MATERIALS, SWEATING MATERIALS, COMPOUND CERAMICS

"Sweating" materials are widely known today. It can be assumed that similar principles for the cooling of rocket combustion chambers have been developed elsewhere, but I should like to mention at this point that my former colleague MEYER-HARTWIG had already devoted intensive studies to these questions 15 years ago. Most significant is the additional boundary layer which, similar to the LEYDENFROST phenomenon, protects the wall from the penetrating heat, cools the boundary layer by evaporation and absorbs heat from the wall. These three single effects result in a remarkable reduction of the surface temperature, which allows the application of sweating materials even at temperatures which would destroy externally cooled solid metals.

At a gas temperature of  $1100^{\circ}\text{C}$  and a flow velocity of 600 to 700 m/sec the surface temperature can be reduced to  $100^{\circ}\text{C}$  applying 0.04 g of coolant per second per  $\text{cm}^2$ . Similar temperatures obtained with air as coolant prove the efficiency of the evaporating veil.

Fig. 15 shows the specific coolant flow rates necessary for higher temperatures.

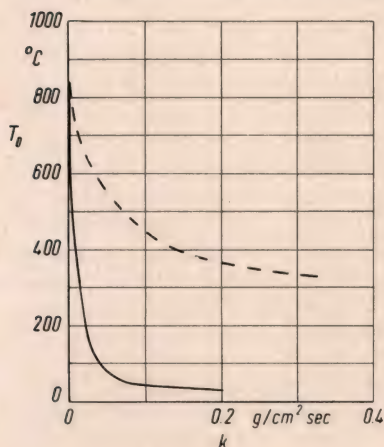


Fig. 15. Surface temperature as a function of specific coolant flow rate

Gas temperature =  $1100^{\circ}\text{C}$ , gas pressure = 24 ata, nozzle diameter = 20 mm, weight rate of gas  $\approx 100 \text{ g/sec}$ , material: copper, coolant: ——— air, — — — water

Powder metal techniques allow the manufacture of parts with sufficient strength and suitable porosity — as for instance exhaust nozzles. Here the nozzle has an inner wall of “sweating” material and a sealing jacket of solid steel. We mainly studied “sweating” materials of copper and iron, but other materials, such as aluminium, can be applied as well. Compared with solid materials the strength of “sweating materials” is reduced roughly in proportion to their porosity. From steel, “sweating” materials can be produced with tensile strengths exceeding  $50 \text{ kg/mm}^2$ .

We started the development of “sweating” materials with non-metallic ceramic materials, but great difficulties arose due to the different thermal expansion coefficients and we had to consider metallic “sweating” materials. As the application of ceramics would offer many advantages, we tried to weld this material to the metal structure. Zones of metal were sintered to the ceramic part, thus forming intermediate layers with increased metal concentrations. Stresses induced by the difference of materials could be reduced gradually by this method. Fig. 16 shows a test rod and nozzle made of compound material.

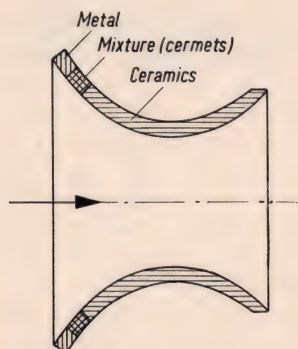


Fig. 16. Test rod and nozzle of compound material

By appropriate further development of compound materials a material should result which will resist high temperatures without any cooling, and which should be specially suitable for gas turbine blades.

# THE DESIGN AND DEVELOPMENT OF THE SOLID-FUEL ROCKET AND ITS PERFORMANCE

HERMANN VÜLLERS \*

The conversion of the solid-fuel rocket from black powder propulsion to smokeless powder propulsion was no easy task, since they are subject to different combustion laws. The control nozzle employed for greater safety in the initial stages of development could only be discarded when, after a more accurate study of the combustion law, a suitable type of powder was eventually found. Some of the more important data necessary to determine the dimensions of a solid-fuel rocket, the power-output obtained and the general trend of development envisaged at the end of the war are described in this paper.

What follows should be prefaced with the remark that it is not intended to give a clear-cut picture of the present state of development of the solid-fuel rocket. This paper is the result of a request that the author should give an overall and retrospective account of the knowledge and experience acquired in this field in the course of ten years' development work at RHEINMETALL-BORSIG. In preparing it the author, who has not worked in this field for the past eleven years, has depended upon the reports of former colleagues \*\*, the available literature, and recollections of the development work carried out at the RHEINMETALL-BORSIG factories in Düsseldorf and Berlin-Marienfelde.

The fact that the solid-fuel rocket was operated on black powder for more than forty years after the invention of smokeless powder is apt to be surprising today. One can read<sup>1</sup> of the laborious efforts devoted to the manufacture of compressed black powder cakes such as were needed by OPEL for launching their rockets in 1928 and 1929. Added to this, there was the uncertainty of whether or not hair-cracks in the powder cakes or fine gaps between powder cake and shell had formed due to vibrations during transport or by variations in temperature during storage — defects which were bound to lead to the destruction of the rocket while burning out. Further, the higher energy content of smokeless powder as compared to black powder, the possibility of manufacturing by up-to-date methods very uniform cakes of any desired shape, weight and calorific value, and the fact that smokeless powder will burn away almost without leaving any residue, were all points in favour of using it in solid-fuel rockets. Apparently the rocket, as old almost as gunpowder itself, needed the modern art of ballistics to give it a new invigorating stimulus, for it

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\*\* In particular, Dipl.-Ing. Helmut Müller is thanked for his co-operation.

was in the Department of Ballistics at RHEINMETALL-BORSIG that the change-over from black powder to smokeless powder was made in 1934<sup>2</sup>. The decisive step was taken when, after several more or less fruitless tests — in the course of which nitroglycerine powder was pressed into a combustion chamber — powder bars were placed in the combustion chamber in such a manner that all surfaces could burn out simultaneously.

Any powder will burn away from the powder surface inward in parallel layers at a rate which, as a rule, depends on the existing pressure. There is, however, a fundamental difference between the combustion of black powder and that of smokeless powder. Because of its porosity, black powder has an extremely high combustion velocity, even under normal atmospheric conditions; it may be as high as 400 m/sec and, if the powder is to be used in rockets, this must be reduced by adding pulverized coal, i.e. the powder must be made "sluggish". Compared to that of black powder, the combustion of smokeless powder is exceedingly slow. Below a certain pressure combustion is even incomplete; nitrous gases are formed. At low pressure, the combustion velocity is approximately 0.005 m/sec, and it increases with increasing pressure. In a gun-barrel, where pressures may rise to 4,000 kg/cm<sup>2</sup>, low-pressure combustion phenomena are of little significance. The rocket, however, which is necessarily of light construction, can only be operated at low combustion-chamber pressures.

In internal ballistics, the law of combustion is normally written, after CHARBONNIER, as follows:

$$(1) \quad dy/dt = A \varphi(y) \psi(p),$$

where

- $y$  = fraction of charge burnt during time  $t$ ,
- $A$  = a constant depending upon the powder,
- $\varphi(y)$  = function of fraction of burnt charge (form function),
- $\psi(p)$  = function of combustion-chamber pressure  $p$ .

When considering the combustion phenomena which occur in the combustion chamber of a rocket, it is, however, convenient to put this law into a different form and to express it as the weight of powder burnt during unit time:

$$(2) \quad W'_p = \gamma S u,$$

where

- $W'_p$  = weight of powder burnt out in unit time,
- $\gamma$  = specific weight of powder (weight per unit volume),
- $S$  = surface area of the powder charge,
- $u$  = linear combustion velocity of powder.

Since the specific weight of the powder is a constant, the weight  $W'_p$  of the gas which develops in the combustion chamber is solely determined by the powder surface area  $S$  and by the combustion velocity  $u$ .

As shown in Fig. 1, the surface area of the powder is not a constant. The left part of this illustration shows a type of single-tube charge and the right, a type of multi-tube charge. As the single-tube charge burns out, its surface area decreases; it is therefore said to be "regressive". As the multi-tube charge burns out, its surface area increases ("progressive") during which process the triangular

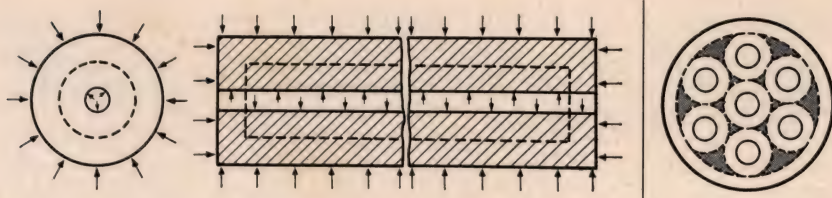


Fig. 1. Behaviour of powder surface during burning  
 Single-tube charge. Surface area decreases (regressive)  
 Multi-tube powder. Surface increases (progressive)  
 ——— Surface when burning starts  
 - - - Surface when burning finishes

residues do not normally burn away completely but are discharged partly unburnt. It is, of course, possible to make a multi-tube charge with a cross-section so designed as to leave no unburnt residues whatever (so-called "profile powder"). For a single-tube charge, the regressivity  $R$  is expressed as follows:

$$(3) \quad R = \frac{S_a - S_e}{S_a} = \frac{2}{l/s + 1},$$

where

$S_a$  = surface area of powder when burning starts,  
 $S_e$  = surface area of powder when burning is completed,  
 $l$  = length of powder tube,  
 $s$  = wall-thickness of powder tube.

The longer the powder tube and the thinner its wall, the smaller will be the regressivity. The charge illustrated in the left part of Fig. 1 has the following dimensions: length  $l = 840$  mm, outer diameter  $d_0 = 130$  mm, inner diameter  $d_1 = 20$  mm. Its regressivity is, therefore,  $R \approx 12\%$ . The seven-tube powder bar shown on the right, on the other hand, has the following dimensions:  $l = 840$  mm,  $d_0 = 170$  mm,  $d_1 = 20$  mm; its progressivity is, accordingly, approximately 40%. There is also the possibility of covering part of the powder surface, either at the beginning or during the whole of the burning process; this will be dealt with later on. It should, however, be stressed once more that the behaviour of the powder surface area is fully under control during the entire burning process. As a rule, one attempts to keep it as nearly constant as possible.

The second factor affecting the quantity of gas developed is the combustion velocity  $u$ , which is chiefly dependent on the pressure in the combustion chamber, the composition of the powder as expressed by its calorific value and, to a minor degree, on the initial temperature of the powder. This last factor will be ignored for the time being. Fig. 2 illustrates the combustion velocity  $u$  as a function of combustion-chamber pressure  $p$  and calorific value  $H$ . In order to compare them to the data established by RHEINMETALL-BORSIG, the data arrived at by MURAOUR are shown as well. According to MURAOUR, the formula should read  $u = a + bp$ , whereas RHEINMETALL-BORSIG — at least for the solid curve shown in Fig. 2 — used the formula  $u = c_1 p^{0.8}$  ( $a$ ,  $b$  and  $c_1$  are constants depending on the powder employed). If we substitute this value in equation (2), we get

$$(4) \quad W'_p = c_2 S p^{0.8}.$$

According to the gas laws, the weight  $W'_N$  of gas flowing through a nozzle during unit time will increase, at supercritical pressure, in proportion to the combustion chamber pressure, i.e.:

$$(5) \quad W'_N = S_N \psi_{\max} p \sqrt{2g/RT},$$

where

- $S_N$  = minimum cross-sectional area of nozzle,
- $\psi_{\max}$  = a factor dependent on adiabatic coefficient  $\kappa$ ,
- $p$  = pressure in combustion chamber,
- $T$  = temperature in combustion chamber,
- $R$  = gas constant.

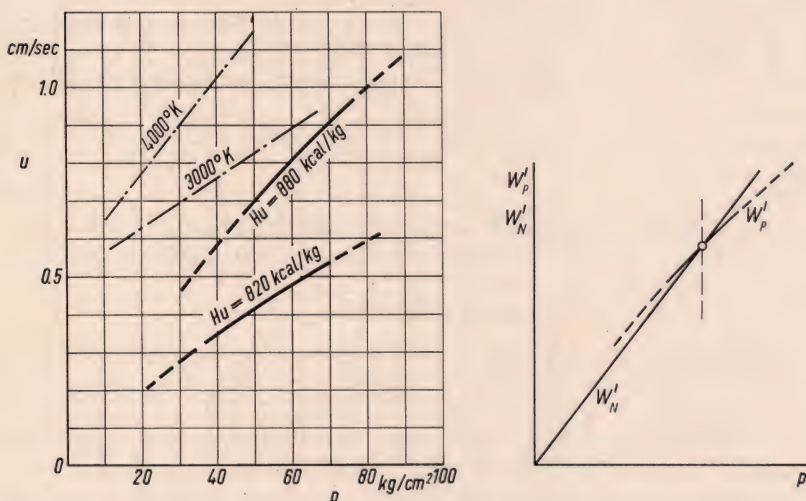


Fig. 2. Combustion velocity  $u$  of smokeless powder as function of pressure  $p$

— According to results of Rheinmetall-Borsig

- - - according to Muraour

Calorific value  $Hu = 880$  and  $820$  kcal/kg

Temperature in combustion chamber =  $4000$  and  $3000^\circ\text{K}$

Fig. 3. Powder combustion rate and gas discharge rate as a function of combustion chamber pressure

- - -  $W'_p$  = Weight of powder burnt in unit time

—  $W'_N$  = Weight of powder discharged in unit time

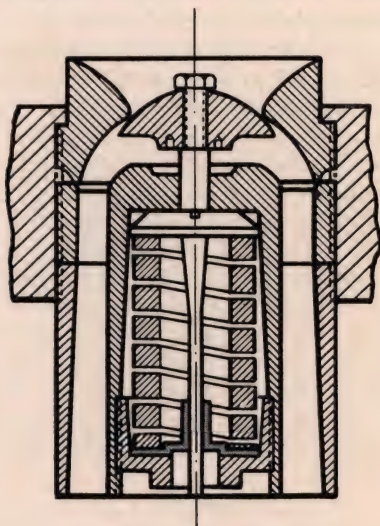
○ = Stable operating point

In Fig. 3, the weight of the powder  $W'_p$  burnt per second according to formula (4), and the weight of the gas  $W'_N$  discharged through the nozzles according to formula (5) are shown as functions of combustion-chamber pressure  $p$ . The point of intersection of the two curves shows the resulting pressure in the combustion chamber. The operating point thus obtained is a stable one for should the pressure in the combustion chamber rise when conditions are near this operating point, the quantity of gas discharged will be

greater than the quantity of gas generated, so that the pressure will again return to normal.

Unfortunately, however, with the majority of powders such stable conditions only occur at comparatively high pressures, viz. approximately 100 kg/cm<sup>2</sup>. Whether the unstable conditions occurring at low pressure are caused by the formation of nitrous gases, by changes in the process of dissociation or by other causes, is an open question. During the tests which were carried out, this instability was very pronounced, in that at very low combustion-chamber pressures the rocket burnt intermittently.

After ignition, the pressure in the combustion chamber returned to zero, gases of a brownish colour were discharged from the nozzle, the pressure rose again, decreased again and so on until, after several intermittent pressure thrusts which grew weaker and weaker, the powder was finally burnt. With a view, therefore, to ensuring with a reasonable degree of certainty the necessary stability in solid-fuel rockets designed for a combustion time of more than two seconds — which in turn made it necessary to operate at very low pressures if a long duration of combustion was to be obtained — RHEINMETALL-BORSIG decided in the earlier stages of development to install at the end of the combustion chamber a control nozzle in addition to the open nozzles. This is shown in Fig. 4. A spring-loaded valve disc of sintered molybdenum, held firmly on its seat by a strong square-edged spring, regulated the pressure; it was set to be half open when the pressure in the combustion chamber was normal, so that it closed when the pressure fell and opened further as the pressure rose.



*Fig. 4. Control nozzle*

Fig. 5 shows the complete propulsion unit of a launching aid for a cargo glider. These devices, which delivered an impulse of 3,000 kg sec, were manufactured in large quantities. The illustration clearly shows the simplicity of construction. The combustion chamber consists of an unmachined seamless tube with a bottom plate welded on to its lower end; the open nozzles and the control nozzle are fitted into the bottom plate. A threaded piece of tube is

welded on to the upper end to take the cap. The tubular charge is kept in position by a grate fitted directly in front of the nozzles and by spacers against the cap. Fitted into the cap there is a hermetically sealed celluloid case containing a priming charge consisting of coarse-grained black powder in the proportion of approximately 10 g for each litre of free combustion space. This



Fig. 5. Solid-fuel rocket with additional control nozzle

priming charge is required to ensure the proper burn-out of the powder bar. Only at very high altitudes has the use of black powder proved unsatisfactory in that there were some instances of misfiring. For use at very high altitudes it was therefore decided to substitute the black powder in the priming charge by a more sensitive powder ("Nudelpulver"). Ignition of the priming charge was effected by an electrically-actuated double-acting ignition plug. This comparatively large priming charge which, however, proved to be indispensable, often led to peak pressures occurring at the start of the combustion process. In order to overcome this, the powder surface had to be partly covered by sticking on strips of paper or cloth which burnt away completely directly after ignition so that the powder surface was then fully uncovered.

It will be seen from equations (4) and (5) that for any given powder the ratio  $S/S_N$ , i.e. the ratio of the surface area of the burning powder to the minimum cross-sectional area of the nozzle, governs the combustion-chamber pressure. This relation, which has been given the name "Klemmung", constitutes one of the most important characteristics for the designer. Its value lay between 420 and 520, depending on the type of powder used. Even during burn-out this figure must not fall below a certain value, because the rocket would then stop burning. This "clinch" principle was also applied to other cross-sections. Within a given powder bar, for cross-sections of the free combustion chamber immediately in front of the grate and in the grate itself the ratio of the powder-surface area to the cross-sectional area, that is to say the "clinch" at any of these points, had not to exceed a certain value, if a ram effect of the gases were to be avoided. At these points a "clinch" of half that corresponding to the cross-sectional area of the nozzle throat appeared to be the most favourable. It was important that the nozzle inlet be well rounded; the cross-sectional area of the nozzle exit had to be about four times that at the throat, and a cone-angle of up to  $35^\circ$  had to be provided.

When ignition first occurs, the powder bar is forced on to the grate by the gas pressure produced in the upper part of the tube near the cap. When the pressure in the combustion chamber has been compensated and a thrust effect has been produced, the powder bar is again forced on to the grate by the acceleration of the rocket. The surface pressure exerted by the powder bar on the grate was less than  $450 \text{ kg/cm}^2$ .

Depending on the available storage accommodation, the initial temperature of the powder bar varied between  $-25^\circ \text{C}$  and  $+40^\circ \text{C}$ . With many types of powder fluctuations of temperature strongly affected the combustion velocity;

at low temperatures the combustion velocity decreased, combustion-chamber pressure and thrust became smaller, and combustion time increased. In order to compensate for these differences, the control nozzle was employed, the preload of the spring being adapted to the temperature of the powder (in accordance with very accurate instructions). Differences between the individual powder charges were also compensated for by means of the control nozzle.

With the constant improvements in the quality of those types of powder in the manufacture of which volatile solvents were not employed (particularly powders in which di-glycol was used), the stability of combustion conditions, even at low pressure, improved to a considerable extent. Further, the combustion velocity of the improved powder was now much less dependent on the powder temperature; in other words, the temperature gradient of the powder had come to be very low. As a result, the control nozzle just described was no longer required. This helped materially to further simplify the construction of solid-fuel rockets and to widen more and more their fields of employment.

At the end of the war, there were rocket propulsion units capable of delivering impulses of up to 40,000 kg sec. The "specific consumption" of powder, i.e. consumption in relation to impulse produced, was approximately 5 g/kg sec, and the specific weight of the complete propulsion unit, ready for use, amounted to approximately 10 g/kg sec.

As an example of a more powerful two-stage solid-fuel rocket, Figs. 6 to 9 show the guided anti-aircraft missile "RHEINTOCHTER". The interesting aerodynamic design of this unit is dealt with in another paper by the author's co-worker Dipl.-Ing. E. MOHR and here only the propulsion of the rocket will be dealt with. The launching stage and the missiles had roughly identical impulses but their combustion times varied considerably. The launching stage — which was also employed for the four-stage long-distance rocket "RHEIN-

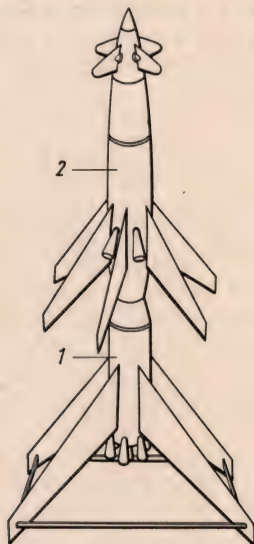


Fig. 6. "Rheintochter". Type R-1 with solid-fuel rockets

1 = Launching rocket: Impulse = 40,000 kg sec, burn-out time = 1.2 sec

2 = Missile rocket: Impulse = 40,000 kg sec, burn-out time = 12 or 60 sec

BOTE" — had a burn-out time of 1.2 sec. The missile had the same impulse of 40,000 kg sec; its burn-out time was originally 12 sec, but when attempts at covering parts of the powder tubes with a fireproof lacquer varnish which prevented the protected part of the powder surface from being set aflame during



Fig. 7.  
"Rheintochter" shortly after  
start



Fig. 8. "Rheintochter"  
before dropping the  
launching stage



Fig. 9. "Rheintochter"  
shortly after dropping  
the launching stage

the entire burn-out period proved successful, it became possible to operate the missile safely with a combustion time of 60 sec. The "RHEINTOCHTER R3" rocket — which used liquid propulsion based on nitric acid and visol — was also equipped with a launching stage consisting of two solid-fuel rockets, one on each side of the missile, which dropped off after burning out. Although liquid-fuel rockets proved to have a better "specific consumption" (consumption in relation to impulse produced), it was planned to continue improving the powder rocket, because of its simple construction and the facts that it was easy to handle and could be stored ready for use. It was intended to operate at still lower pressures (as low as 20 kg/cm<sup>2</sup>), in order to be able to further reduce the weight of the empty rocket. In this connection, the law of combustion was to be further investigated in close co-operation with scientists in the fields of both powder chemistry and thermodynamics. The influence of stabilizers on burn-out velocity and temperature gradient had to be investigated. It was planned to increase the specific weights of the powder (propellant weight per unit volume) and to increase its calorific value as far as possible so as to improve the weight-ratio of the rocket. It was intended to construct propulsion units having a very long combustion time, in which the powder was to be introduced into the combustion chamber cartridge by cartridge from a pressure-free storage chamber in order to reduce the size and thereby the weight of the combustion chamber.

A study of the more recent literature <sup>7</sup> will reveal that further development work in the field of solid-fuel rockets has been carried out. Details of the results obtained and whether or not this more recent research work has had any influence on the design of rocket propulsion is not known.

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## DISCUSSION

Mr. A. R. WEYL (Dunstable, Beds.): Has there been any evidence of irregular burning or sudden rise of pressure because of heat radiation through grain transparency, i. e. premature decomposition of deeper layers in advance of the burning surface? If so, what remedies have been adopted? What measures have been taken to cope, in operational use, with the influence of initial grain temperature upon the burning pressure?

Dr.-Ing. H. VÜLLERS: A method of compensating for the unstable combustion of the powder during the operation in the beginning was to use RHEINMETALL governor valve. As already mentioned, it was no longer necessary when the newer, highly efficient powders were used.

Prof. Dr.-Ing. E. SCHMIDT (Munich): It may be of interest that in my former Institute for Aircraft Research at Braunschweig-Völkenrode the problem of unstable combustion was also studied by my collaborator G. DAMKÖHLER. He got stable operation at rather low pressures — down to about 10 atm — by using smokeless powders with uria nitrate ( $\text{CONH}_2\text{NO}_3$ ) as the main component. As uria nitrate has a surplus of oxygen, other smokeless powders having less oxygen were added.

Dr.-Ing. H. VÜLLERS: This remark of yours stresses what I said at the end, viz. that a lot can be done by further development of the powder.

Dipl.-Ing. E. MOHR (Wuppertal-Vohwinkel): I would like to add to Dr. VÜLLER's lecture a few words on the aerodynamic development of the RHEINTOCHTER.

In 1942, RHEINMETALL-BORSIG received an order for the development of a solid-fuel anti-aircraft rocket with a ceiling of approximately 8 km. The device was to operate in two stages. At that time, the maximum combustion time obtainable with powder combustion did not exceed 10 to 12 sec. If the ceiling called for was ever to be reached, the rocket had to be a supersonic one. There were no great objections to these speeds, since projectiles, winged mines and other flying bodies made by RHEINMETALL-BORSIG to a large extent already operated in the supersonic range; but nevertheless the RHEINTOCHTER was to be constructed in such a way that no difficulties in the aerodynamic development were to be expected. The construction is shown above in Dr. VÜLLER's paper (Fig. 6) and had the following details: The control surfaces were located at the front of the body, the wings were near the centre of gravity and therefore

more to the rear. This arrangement ensured that the wings as well as the control surfaces at the forward end remained (even at supersonic speeds) in a nearly undisturbed air flow.

In aeronautical language this type of construction is referred to as the duck system; however, the RHEINTOCHTER was of cruciform construction with 6 rotationally-symmetrical wings. The control surfaces were mounted in pairs on transverse axes and were moved as a whole, i.e. there was no subdivision into rigid fins and movable surfaces. The control surfaces were moved continuously by their respective motors and a later step in the development provided for a pneumatic two-step setting to right and left or up and down. The position of the axis of rotation resulted from wind tunnel measurements and was chosen so that the hinge moments were a minimum. In the subsonic range these aerodynamic moments increased the angle of incidence, whereas in the supersonic range they decreased it. The position of the wings relative to the centre of gravity was also determined by wind tunnel experiments. All these measurements were made in Prof. WALCHNER's supersonic wind tunnel in Göttingen; those in the subsonic range were carried out in the wind tunnel of the Aachen Technical University. The results were to a great extent confirmed by test flights with models made to a scale of 1:2.5. The wings were delta shaped with an angle of 45°.

The booster was dropped after a combustion time of 1.2 sec; it was fixed at the tail of the rocket and stabilized by four large plywood wings. It should be added that the device was not stabilized in roll; instead the rocket, depending upon its constructional dissymmetry, rotated about its longitudinal axis and divided the incoming steering signal on a gyroscope, which carried four potentiometers, between the two steering axes. Full deflection of the control surface resulted in an angle of incidence of about 90°. The aerodynamic restoring forces were so strong that even with large starting errors the missile could be quickly brought on to its trajectory. Model tests on the controls were carried out in Prof. FISCHER's Institute. Movement of the centre of pressure on passing from the subsonic to the supersonic range was slow enough, although the exact figures cannot be recalled. The fact that TELEFUNKEN had carried out most of the radio-navigation tests on the RHEINTOCHTER also supports the statement that aerodynamic difficulties had largely been overcome. At the end of the war, one hundred R-1 devices (with solid-fuel motors) and roughly six R-3 devices (with liquid fuel motors and two solid-fuel boosters mounted on the sides) had been tested.

# DEVELOPMENT OF HYDROGEN PEROXIDE ROCKETS IN GERMANY

HELLMUTH WALTER \*

## 1. THE SUITABILITY OF HYDROGEN PEROXIDE FOR PROPULSION PURPOSES

The development of hydrogen peroxide propulsion in Germany dated, broadly speaking, from 1935 till 1945. During the two years preceding this period, the properties of  $\text{H}_2\text{O}_2$  were tested in collaboration with the ELEKTRO-CHEMISCHE WERKE at Munich, its suitability was examined, its concentration was increased up to about 80%, and decomposition tests were carried out. As a result of these experiments, it was decided that  $\text{H}_2\text{O}_2$  is suitable for propulsion purposes, at least as far as this could be said without comprehensive practical experience. At that time the relevant departments of the German Armed Forces, in particular the German Naval High Command, were interested in the use of  $\text{H}_2\text{O}_2$  for the propulsion of submarines, so that it was decided to carry out experiments in actual practice.

Certain circumstances favoured this decision. From 1930 till 1934 the author had, in co-operation with the FRIEDRICH KRUPP GERMANIAWERFT at Kiel, developed a gas-turbine plant on behalf of the German Naval High Command. Without going into detail it may be stated here that, with compressor- and turbine-efficiencies of 80% each, a maximum total efficiency of only 20% could be attained even with the use of a heat-exchanger. This was considered inadequate, particularly in view of the satisfactory results which the introduction of high-pressure superheated steam aboard warships appeared to promise. The project was, therefore, dropped. But it did provide the opportunity of using certain parts of the gas-turbine plant for an  $\text{H}_2\text{O}_2$  propulsion device.

This plant, which in 1936 had an output of 4000 HP already contained all the fundamental parts which are required in a rocket propulsion unit, i.e.  $\text{H}_2\text{O}_2$ - and fuel pumps, flow-regulator,  $\text{H}_2\text{O}_2$ -decomposer and combustion-chamber with cooling jacket.

The  $\text{H}_2\text{O}_2$ , concentrated to 80%, was conveyed by a pump through the flow-regulator to the decomposer-chamber where, in a bed formed by metal or porcelain rings covered with a layer of manganite and caustic potash, it was decomposed by catalytic reaction into a steam-oxygen mixture. This was then burnt in the combustion-chamber by adding fuel oil. The combustion-chamber was cooled with water which was injected into the hot gases at its rear end, cooling the gases to a temperature which would not damage the turbine.

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The propellant gases generated in this plant did not expand in a thrust nozzle — as would be the case in a rocket propulsion unit — but in a large number of turbine admission channels. This difference, however, was like so many others as e.g. overall dimensions and weight, only a matter of expediency. The principle realised in this prototype plant was the target continually aimed at during the following years of development and research, yet specific tasks and circumstances, in particular considerations of safety, forced us to advance only step by step, and even to deviate from the line which was regarded as desirable, until finally it was again possible to approach the ideal.

At about the same time the first stand-trials with rockets had been carried out, and thrusts of up to 1000 kg were obtained for several seconds.  $\text{H}_2\text{O}_2$ , concentrated to 80%, was forced by compressed air through a decomposer fitted with perforated metal sheets which were covered with a paste-like catalytic layer similar to the one used in the turbine plant. With a reaction-chamber volume of 3 litres, the mass-flow amounted to about 10 kg/sec. Decomposition was spontaneous and complete. This proved that the reaction could be performed in a chamber sufficiently small to be suitable for rocket propulsion. This result encouraged the German Air Ministry to carry out a trial with an auxiliary rocket propulsion unit, of 130 kg maximum thrust, fitted to a He 52. This unit which could be controlled, was capable of maintaining its maximum thrust for 45 sec. Flight tests were carried out in January 1937; they were successful and free from accidents.

Thus, the single-component procedure, based only upon the energy obtained from the decomposition of the  $\text{H}_2\text{O}_2$ , was found to be suitable in principle. Owing to its low maximum temperature of  $500^\circ$  to  $600^\circ\text{C}$ , it is also called the "cold procedure" contrary to the "hot procedure", where also the oxygen freed by decomposition is burnt together with fuel in the reaction. The practicability of the "hot procedure" was proved by injecting additional petroleum into a "cold" unit in operation; the combustion reaction was spontaneous.

Thus, the use of  $\text{H}_2\text{O}_2$  appeared to be very advantageous indeed in all cases where a maximum power-output was required for a short while, for the smallest possible weight of propulsion unit, or where atmospheric oxygen was not available for combustion.

## 2. PROPERTIES OF HYDROGEN PEROXIDE

$\text{H}_2\text{O}_2$  is a clear water-like liquid of a specific gravity of 1.46 and can be mixed with water in any proportion. Its absolute viscosity is about 33% higher than that of water. On decomposition, 690 kcal and 0.47 kg oxygen per kg  $\text{H}_2\text{O}_2$  are released. The decomposition temperature is  $950^\circ\text{C}$ .

Fig. 1 shows how decomposition temperature, specific gravity and other variables depend on the  $\text{H}_2\text{O}_2$  concentration, based on an initial temperature of  $0^\circ\text{C}$ . It can be seen that e.g. the decomposition products of 15%  $\text{H}_2\text{O}_2$  will attain a temperature of about  $80^\circ\text{C}$  on decomposition. If the concentration exceeds 65%, a mixture of superheated steam and oxygen only is obtained, the temperature of which finally amounts to  $950^\circ\text{C}$  in the case of a 100% concentrated  $\text{H}_2\text{O}_2$ .

The decomposition temperature increases with increasing pressure, the difference, however, becoming less marked with higher concentrations and

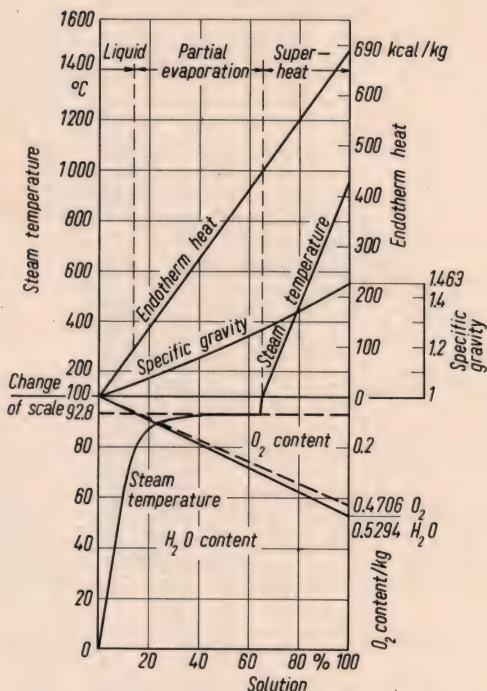


Fig. 1. Relation between  $T$ -material and concentration at  $p = 1$  ata and  $t_0 = 0^\circ\text{C}$

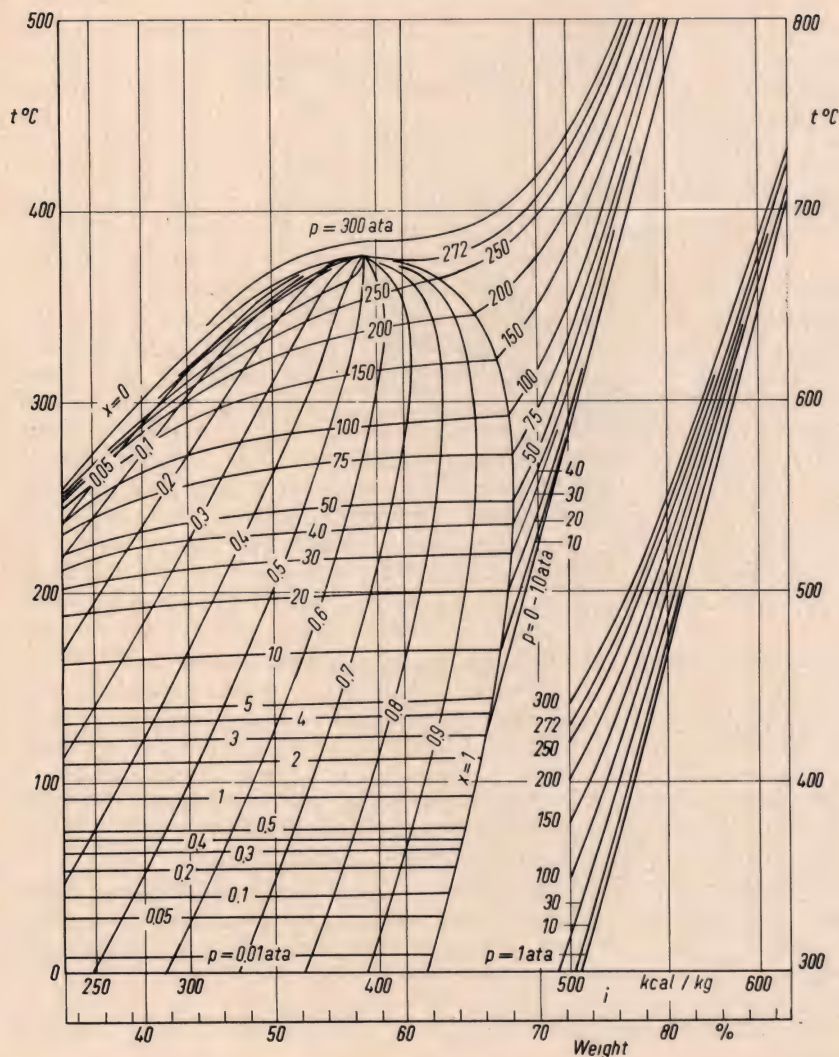
temperatures. As will be seen from Fig. 2, the difference between 1 ata and 300 ata (ata = absolute atmospheric pressure) is  $200^\circ\text{C}$  when using 70%  $\text{H}_2\text{O}_2$ , but is only  $25^\circ\text{C}$ , when 90%  $\text{H}_2\text{O}_2$  is used.

By burning  $\text{H}_2\text{O}_2$  with hydrocarbons in stoichiometrical proportion, about twice the decomposition heat is produced in addition. If  $\text{H}_2\text{O}_2$  of 100% concentration could be used, altogether 2000 kcal per kg  $\text{H}_2\text{O}_2$  would be produced, and a combustion temperature of about  $2500^\circ\text{C}$  would be reached. The practical limit, however, is about 90%. Up to the end of the war, however, concentrations of more than 85 or 86% were not used for the following reasons:

1. Combustion temperatures became too high. Since high alloy steels were not available, only low alloy steels could be used for the combustion-chambers.
2. The freezing point of  $\text{H}_2\text{O}_2$  of 85% concentration is  $-16^\circ\text{C}$  and increases with increasing concentration.
3. Experiments had shown that  $\text{H}_2\text{O}_2$  concentrated to more than 87% is liable to detonate.

In actual practice, no difference was noted between  $\text{H}_2\text{O}_2$  of 80% and of 85% concentration.

The freezing point of  $\text{H}_2\text{O}_2$  depends largely on the concentration. With concentrations between 50% and 60%, the freezing point lies below  $-50^\circ\text{C}$ , whereas with decreasing or increasing concentration it goes up in direct proportion to  $0^\circ$  and to  $-2^\circ\text{C}$  respectively. It can be seen from Fig. 3 that

Fig. 2.  $t, i$ -diagram for T-material

Decomposition temperature and moisture content at various pressures;  $i$  = enthalpy,  $t$  = temperature, initial temperature  $t_0 = 0^\circ\text{C}$

the freezing point of the most preferred concentration of 85 % is about  $-15^\circ\text{C}$ . This did not lead to difficulties in practice; yet in one instance, i. e. in the case of the aerial torpedo, this characteristic was taken into consideration and a concentration of 59 % was used. It is self evident that the freezing points of the decomposing solutions and burning fuels must be just as low.

In this connection attention may be drawn to the rather high dependence of the specific volume on the temperature, which amounts to about 5 % in the temperature range from  $-20^\circ$  to  $+50^\circ\text{C}$ . This property is rather more important than it may appear at first glance. Overflowing of  $\text{H}_2\text{O}_2$  must be

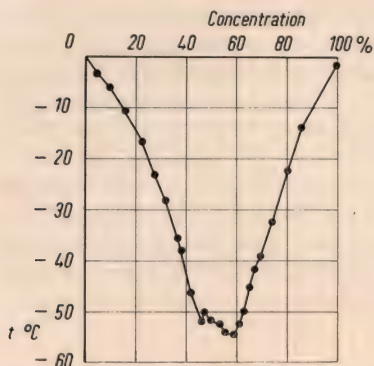


Fig. 3. Freezing points for T-material

avoided due to the rather disagreeable effects it produces, particularly in closed rooms.

To determine whether and when  $\text{H}_2\text{O}_2$  is likely to detonate, detonation tests with  $\text{H}_2\text{O}_2$ -ethylalcohol-solutions were carried out, the results of which are set down in Fig. 4. The diagram shows that with the solution containing 0% alcohol,

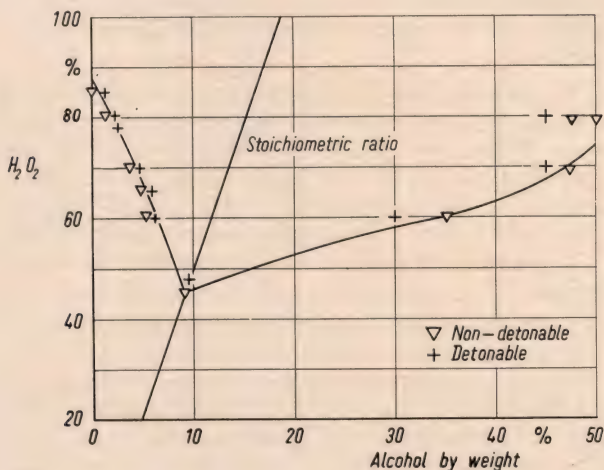


Fig. 4. Chart of detonation border-line for the mixture T-material — alcohol

$\text{H}_2\text{O}_2$  will become explosive when concentrated to 87%. In previous tests carried out in 1934 by the CHEMISCH-TECHNISCHE REICHSANSTALT at Berlin-Halensee, no additional effects were observed, when  $\text{H}_2\text{O}_2$  was used instead of water together with a powerful detonator (lead acid). Furthermore, it was found that high pressure — contrary to its application to explosives — had a retarding effect on an already initiated decomposition process. Occasionally, decomposition was stopped completely. Thus it was established that neither shock nor pressure could cause  $\text{H}_2\text{O}_2$  concentrated up to 87% to disintegrate

suddenly. Later tests have shown<sup>1</sup> that even with a 90%  $\text{H}_2\text{O}_2$  concentration no propagating detonation wave was met with, so that these days a 90% concentration is still considered as admissible. With large initial charges, however, it may happen that a highly concentrated solution of  $\text{H}_2\text{O}_2$  will "go off" together with the charge; this is partly due to the resulting heat causing evaporation spots and thermal decomposition. At least this was the subjective impression of the author assisting at a test where 500 kg Hexogen were detonated in a solution consisting of 5000 kg  $\text{H}_2\text{O}_2$  concentrated to 85%. Mixtures or solutions of fuels with  $\text{H}_2\text{O}_2$ , whether intentional or not, should however be avoided. For further details, also for questions of handling and storage, reference should be made to earlier publications<sup>2,3</sup>.

### 3. FUNDAMENTAL PROBLEMS OF HYDROGEN PEROXIDE PROPULSION SETS

Next to the problem of storage the ever changing problems associated with the conveyance of fuel had to be solved. As mentioned before, pumps or compressed air were used for this purpose at first. Later on, delivery methods were developed in which the delivery gas-pressure was generated by the fuel itself as e.g. by decomposition of  $\text{H}_2\text{O}_2$ , resulting in some cases in the air compressors becoming superfluous. It may be stated here that an inert gas has never been used for the conveyance of petroleum, but invariably air. Explosions have never occurred.

One of the outstanding advantages of hydrogen peroxide is the possibility of using it as a single-component-system. By surface catalysis, a propellant gas is produced whose temperature is only dependent upon the concentration. An additional temperature-regulation or proportioning as with a bi-component system will, therefore, not be required. The  $\text{H}_2\text{O}_2$  method has recently been described in full detail by EMIL KRUSKA, the merited assistant of the author for many years<sup>4</sup>. Therefore, only some self-delivery methods tested in practice will be dealt with here.

First, however, it should be mentioned that  $\text{H}_2\text{O}_2$ -containers must be aerated continually, though the pressure within the container does not actually affect the  $\text{H}_2\text{O}_2$ . A large number of torpedo-containers holding approximately 200 litres were subjected to storage tests under tropical conditions. After storage for 6 months at a temperature of 50°C, the concentration had hardly changed.  $\text{H}_2\text{O}_2$  of 80% concentration, produced in Germany in 1944, was used in the United Kingdom 4 years later and was found to still have its original concentration though it had been several times re-filled in transport. Rocket-driven planes, auxiliary starting devices etc. may, therefore, remain fuelled for several months. The problem becomes more complicated with containers holding less than 20 litres. Here a certain limit is imposed. In addition, due to the necessary aerating,  $\text{H}_2\text{O}_2$ -appliances cannot be as easily handled and transported as e.g. powder-rockets.

In the following review of delivery methods some self-delivery installations will be explained in detail. Fig. 5a shows the simplest case. After opening a valve  $\text{H}_2\text{O}_2$  flows into a decomposer installed below. The steam — oxygen mixture then flows partly — to the right — to its proper use, and partly — to

the left — back to the  $\text{H}_2\text{O}_2$  pressure tank. The mass-flow is determined by the static head, by the resistance of the tube and of the decomposer and by the opening of the valve. When a differential piston as shown in Fig. 5 b is applied, the rate of flow will be increased and becomes independent of the force of gravity. The small cylinder is filled with  $\text{H}_2\text{O}_2$ , whereas the gas pressure will act on the large piston after being released by a small force. A propellant gas plant for a catapult worked according to this principle and delivered 100 kg  $\text{H}_2\text{O}_2$  per second for a short period. Fig. 5 c and 5 d show the lay-out of two pressure-gas delivery plants for bi-component systems, 5 c with simple differential piston and injection delivery, 5 d with twin differential piston and hot gas delivery.

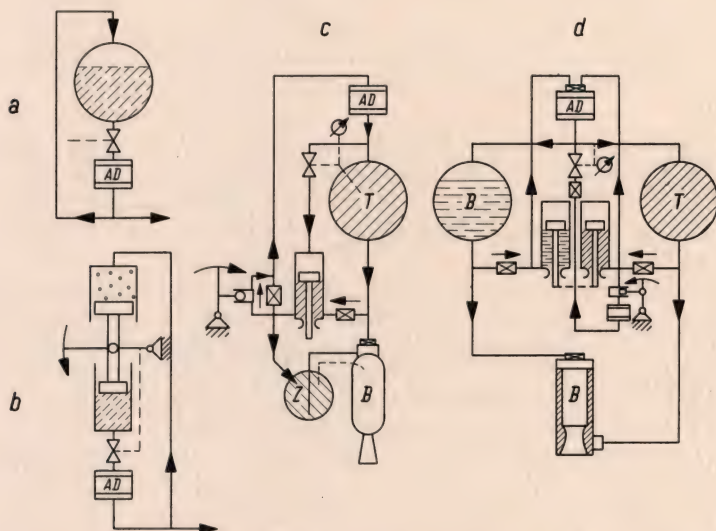


Fig. 5. Pressure-gas generation from *T*-material (power unit for short operation time)  
AD = Auxiliary decomposer

When turbine-driven pumps are employed (Fig. 6 a), starting is e.g. effected by an electric starter motor of small power fitted with an overriding clutch. As soon as the turbine pump begins delivery, a partial flow will pass through a decomposer. The gas generated here starts in its turn to drive the turbine and will bring the rotor to the number of revolutions required in a few seconds. Control is effected by throttling and by changing the rotational speed.

In Fig. 6 b starting is effected by an air-cushion in a small  $\text{H}_2\text{O}_2$  pressure tank. The tank is refilled automatically once the turbine-pump has begun to work. Control is effected on similar lines as above.

As development progressed, it was found that the air-cushion could be eliminated (Fig. 6 c). A static height of 20 to 30 cm between auxiliary tank and auxiliary decomposer was found to be all that was needed for starting.

Turbine pumps, main and auxiliary decomposers and control equipment have already been described, as well as several types of combustion chamber<sup>4</sup>.

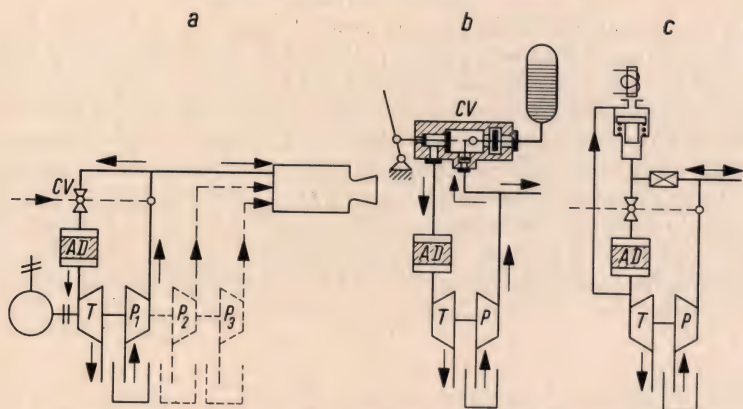


Fig. 6. Controllable power unit for long time operation (Walter auxiliary power unit)  
AD = Auxiliary decomposer, CV = Control valve

The  $\text{H}_2\text{O}_2$  power plants developed by using the auxiliary apparatus mentioned above were classified into "cold" plants, i. e. operating on decomposition only, and "hot" plants, i. e. plants, where the oxygen was also burned. Decomposition was effected either by injection of a permanganate solution or else by solid catalytic agents.

When using the first procedure,  $\text{H}_2\text{O}_2$  was reduced, whilst the permanganate decomposed into hydroxide and manganese dioxide which had a strong catalytic effect. The ratio between the permanganate solution of specific gravity of 1.5 and  $\text{H}_2\text{O}_2$  of 80% concentration was approximately between 1:30 and 1:15, whilst the space required for decomposition amounted to about one litre for each kg  $\text{H}_2\text{O}_2$ . The space required for the use of solid catalytic agents was about five times that figure. With 80%  $\text{H}_2\text{O}_2$  and by a 20-fold expansion the specific thrust of "cold" power units amounted to 111 kg/kg and, accordingly, the specific consumption amounted to 0.009 kg/kg. The "cold" procedure was considered to be extremely safe. Though it was used thousands of times, no accidents occurred. Normally, the units were adjusted with compressed air and water to obtain the desired ratios. Units delivered in mass production from factories were rendered "alive" only prior to their actual employment at the front.

#### 4. "COLD" POWER UNITS

In Fig. 7 is shown a cross-section of the auxiliary take-off rocket 109—500 A. This was one of the most important "cold" power units, and it was widely used in practice. The unit which burnt for 30 sec delivered a thrust of 500 kg. It was dropped by parachute and re-employed. In the summer of 1937, these units were attached to a Do 18 and to a He 111. Also the Ju 88 was normally equipped with fittings for auxiliary take-off devices. During the war about 3000 operational flights were performed with these units all over Europe without accident.

Fig. 8 shows the lay-out of another unit 109—507 used in the propulsion of glider bombs. The arrangement of the containers for  $\text{H}_2\text{O}_2$  ("T-material") and permanganate ("Z-material"), as well as the arrangement of the air-bottles

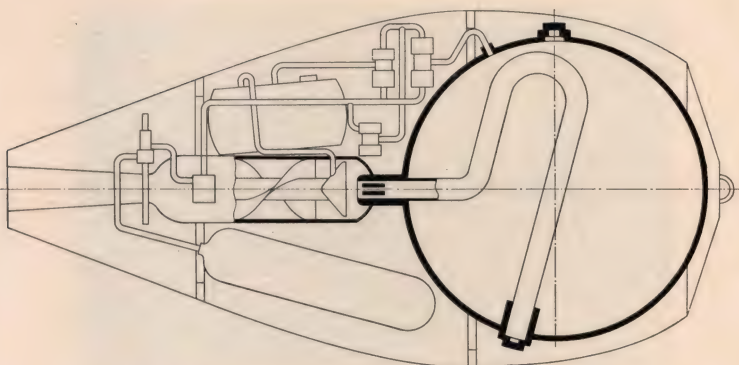


Fig. 7. Assisted take-off unit 109—500

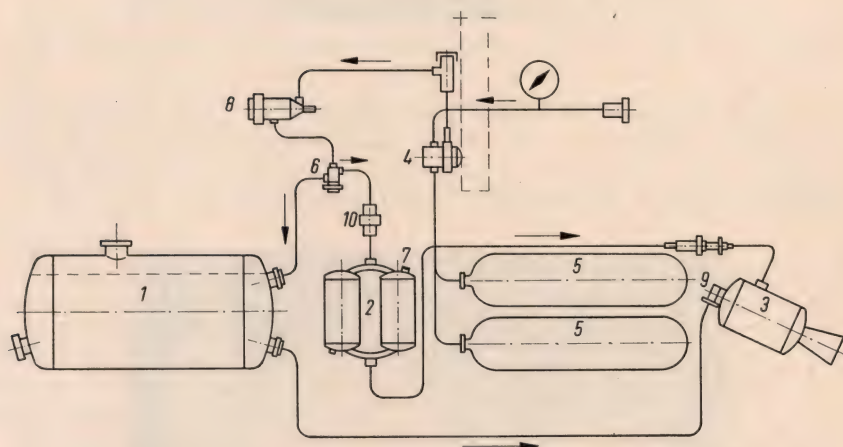


Fig. 8. Lay-out of the power unit 109—507 for gliding bombs

- |                              |                              |
|------------------------------|------------------------------|
| 1 = Container for T-material | 2 = Container for Z-material |
| 3 = Decomposer               | 4 = Igniter                  |
| 5 = Air bottle               | 6 = Vent for T-material      |
| 7 = Vent for Z-material      | 8 = Pressure reducing valve  |
| 9 = Injection valve          | 10 = Non-return valve        |

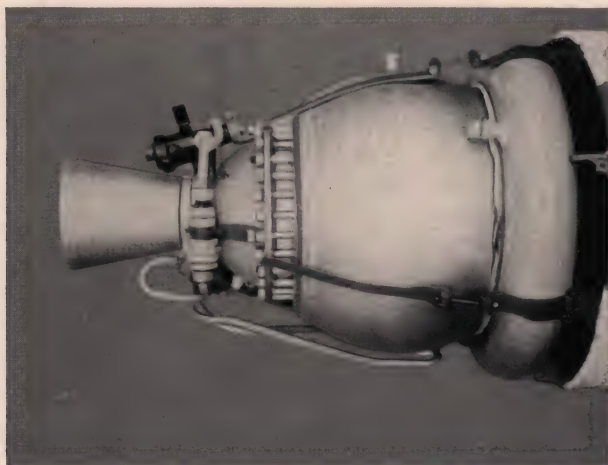
and of the decomposer with nozzle can be gathered from the drawing. Activation was effected by a small explosive charge which was electrically ignited and which penetrated a diaphragm. The air-pressure was regulated by a reducing-valve. Thrust amounted to 600 kg and decreased during the operational time of 10 sec to 400 kg. This unit was used for propelling the tele-guided glider bomb Hs 293 developed by Prof. WAGNER, which was suspended beneath the carrier-aeroplane (Fig. 9).

A similar unit was used for propelling the tele-guided aerial torpedo BV 143. The aerial torpedo was fitted beneath the carrier-plane (e.g. He 111) and was given its direction prior to release close to the water surface by mechanical means.



*Fig. 9. Power unit 109—507 in gliding bomb Hs 293 mounted underneath carrier aircraft*

It may be mentioned here that rockets working on the "cold" principle were also used for retarding the fall of sea mines which, as distinct from those dropped by parachute, were aimed at special sites. A rocket of this type having a high thrust and a short combustion time is shown in Fig. 10.



*Fig. 10. Rocket with short burning time for decelerating the fall of sea mines*

Glider bombs were not only dropped from carrier-planes, but also launched from catapults. These were driven partly by compressed air, partly by the "cold" procedure. E.g. in the split-tube catapult shown in Fig. 11, which was used for the aerial torpedo BV 143 mentioned above; for the first time the cylinder itself was used as a decomposition chamber. The catapult operated, without a cable, with a sealing strip running through the piston and closing the long slot of the tube. The piston was ejected and normally re-used.



*Fig. 11. Split-tube catapults for V-1 missiles*

The catapult type used for the V-1 could be assembled from elements 5 m long in a few hours by means of a hand-crane running along the launching track. The launching velocity was about 105 m/sec and could be increased to 250 m/sec with smaller loads without any trouble. With a launching velocity of 105 m/sec, the time of operation was about 1 sec, when about 100 kg  $\text{H}_2\text{O}_2$  were decomposed.

Propulsion of a manned aeroplane by rockets alone was carried out for the first time in summer 1939; the plane used was the He 176, specially built for this purpose.

For the tailless aeroplane Me 163 A, designed by LIPPISCH, the propulsion unit R 11, 203 B which had a maximum thrust of 750 kg was developed; the lay-out is shown in Fig. 12. In this unit the control of  $\text{H}_2\text{O}_2$  and "Z-material" in the ratio of about 100:750 was effected by a pressure-reducing regulator. Due to the different quantities, the difference in pump characteristics was compensated by a pressure balance.

The decomposer was of tubular shape to enable the centre of gravity of the unit to be placed as close to the front end as possible. It consisted of a mixer and of an extended decomposer jet in which decomposition took place gradually, whilst the velocity of the gas increased steadily. The difference of efficiency compared with the conventional type of decomposer was negligible.

Certainly several thousand flights were made with the Me 163 A training aircraft. In 1942, this plane surpassed for the first time in history the speed of 1000 km/h in horizontal flight. It climbed to a height of 10,000 m in less than three minutes.

It may be of interest to note that about a year before the Me 163 attained the highest speed ever reached in the air at that time, the experimental submarine V 80 set a world-record for underwater travel with a speed of 26.5 knots. It also had a "cold" controllable power unit, but catalytic surface decomposition. The driving unit consisted of fuel pump, decomposer and turbine delivering 2000 HP at 20,000 revolutions and the necessary gear mechanism. More than 80 test runs were undertaken with this boat and valuable experience was gained. Depth control did on the whole not offer any serious difficulties.

Owing to its simplicity and working reliability, the "cold" procedure was used, apart from the rockets and power units mentioned above, for a large

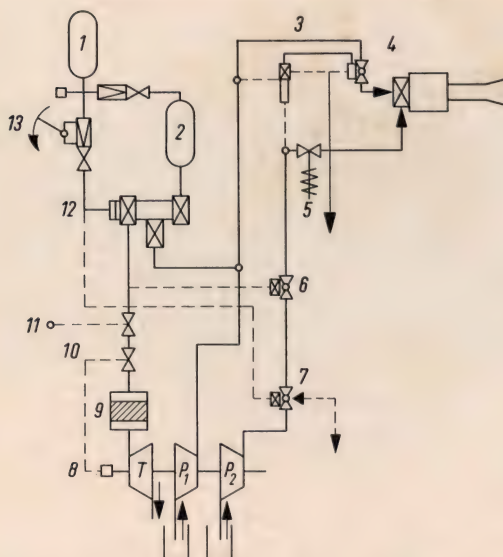


Fig. 12. Controllable "cold" power unit R 11—203 b for long time operation

- |                                  |                             |
|----------------------------------|-----------------------------|
| 1 = Air bottle                   | 2 = T-material for start-up |
| 3 = Pressure balance             | 4 = Valve for T-material    |
| 5 = Control valve for Z-material | 6 = Valve for Z-material    |
| 7 = Drain                        | 8 = Turbo pumps             |
| 9 = Auxiliary decomposer         | 10 = Trip                   |
| 11 = Emergency trip              | 12 = Main control valve     |
| 13 = Pressure reducer            |                             |

number of other purposes as e.g. for the turbine pump drive of the V-2, for the V-2 model-rockets, for power units of winged speed boats, for wing boundary layer control so as to increase the lift and — last but not least — even for an  $\text{H}_2\text{O}_2$  cannon.

The largest rocket ever built to work on to the "cold" principle was envisaged for use in a special landing craft. It developed a thrust of about 25,000 kg. The landing craft, carrying an explosive charge of 5 tons, was brought near to the shore and, after starting its rocket drive, pushed on its skids several hundred metres inland.

## 5. "HOT" POWER UNITS

It was mentioned at the beginning that the first  $\text{H}_2\text{O}_2$  power unit ever developed worked on the combustion principle. The propellant was decomposed by surface catalysis. In the beginning with 0.1 kg/sec of propellant being put through, the decomposer volume was about one litre and later on was gradually decreased to half this figure, yet it was still too large to be suitable as a propulsion unit for rockets. Also the decomposer was too heavy. It was found that ignition was obtained immediately when fuel was injected into a unit which worked "cold" with "Z-material". Combustion continued unaffected

when the injection of "Z-material" was stopped. This procedure was used for torpedo propulsion, whereas in rocket propulsion the "Z-material" was injected throughout the whole time of operation for safety reasons.

Tests in the use of powder units for ignition purposes were not always successful and even resulted in explosions. Initially, therefore, ignition was effected by "Z-material".

This procedure required, even though only in small quantities, one component more than appeared desirable and rather complicated the plant. When, therefore, LUTZ, HOFMANN and NOEGGERATH in 1939 suggested the use of hydracine hydrate and mixtures of this component with methanol, the series 109—509 working on this system was developed. Finally, when the catalysts for surface decomposition had been further improved and as the decomposers became smaller and lighter, an attempt was made to adapt the procedure which had proved so successful from the beginning for submarines, to aircraft power units. Contrary to the previous procedure, self-ignition was thought to be desirable for aircraft propulsion; hitherto, high-tension ignition had been required for starting. Self-ignition was now effected by pre-heating  $\text{H}_2\text{O}_2$  of 80% concentration by auxiliary steam. Decomposition temperature in this case rises by  $1.5^\circ$  for each degree of pre-heating.

But at that time it was not possible to obtain, by surface catalysis in a sufficiently small decomposer, the mass-flow of 7.5 kg/sec of  $\text{H}_2\text{O}_2$  required for interceptor power units. Therefore, only a part of the flow was pre-decomposed and, after self-ignition, was burnt with a hydrocarbon in a pre-combustion chamber. Higher thrusts were obtained by direct injection of fuels into the combustion chamber. Dangerous agglomerations of liquid fuel were avoided due to the continuous flow of gas. Thus we were one step nearer to the desired goal, i. e. to working with pre-decomposition.

Of the three methods employed for the combustion of  $\text{H}_2\text{O}_2$ , the one working with pre-decomposition is the most reliable and the safest and, unless the single-component procedure is used, should always be employed.

Combustion will reduce the specific consumption of the rocket from about 9 g to 4.5 g/kg thrust per second.

Comparison of the temperature gradients in a fully approved self-igniting "M-material" (consisting of 57% methanol, 30% hydracine hydrate and 13% water) with a hydrocarbon (85% C and 15%  $\text{H}_2$ ) shows the following: When being burnt with 80%  $\text{H}_2\text{O}_2$ , this "M-material" develops a maximum temperature of  $1950^\circ\text{C}$ , whereas hydrocarbon will, under the same conditions, develop a maximum temperature of  $2200^\circ\text{C}$ . In the latter case the temperature gradient is about 7% higher. Combustion of a hydrocarbon of the same composition with 90% nitric acid develops, however, a maximum temperature which is  $400^\circ\text{C}$  higher, although the enthalpy is almost identical with that of 80%  $\text{H}_2\text{O}_2$ .

Now some power units working on the combustion principle will be described in detail.

The take-off aid 109—501 (Fig. 13) developed a thrust of 1500 kg for 28 sec. The unit could be switched off. Nozzle and combustion chamber were cooled with  $\text{H}_2\text{O}_2$ . "Z-material",  $\text{H}_2\text{O}_2$  and fuel were automatically cut in in this order and were cut off again, when required, in the reverse order. To ensure complete combustion, an excess of "Z-material" was available. Delivery

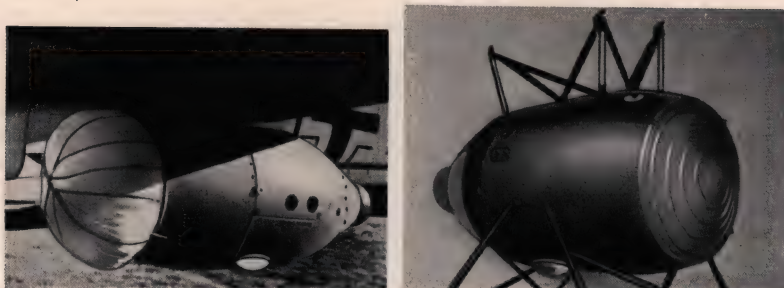


Fig. 13. Assisted take-off unit 109—501

was effected by compressed air. Only a small number of this unit, which was not tested operationally, were built. It was, however, frequently used for trial purposes, as e.g. for the jet-bomber Ju 287 and for the tele-guided projectile "ENZIAN".

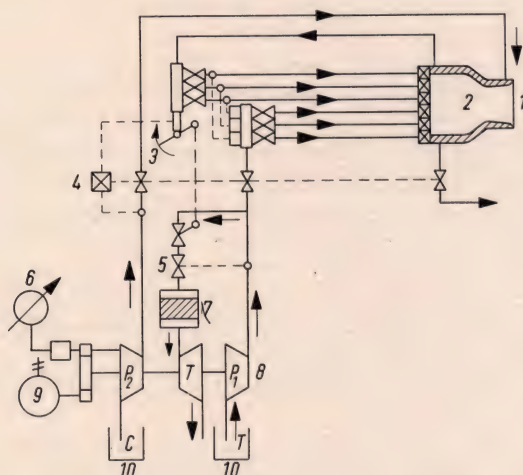


Fig. 14. Controllable power unit 201—509 for long time operation

- |                          |                        |
|--------------------------|------------------------|
| 1 = Nozzle               | 2 = Combustion chamber |
| 3 = Control lever        | 4 = Shut-off valve     |
| 5 = Control valve        | 6 = Tachometer         |
| 7 = Auxiliary decomposer | 8 = Turbo pump         |
| 9 = Starter              | 10 = Container         |

The lay-out of the controllable power unit 201—509, having a thrust of 1700 kg, is shown in Fig. 14. This unit worked with  $H_2O_2$  and "M-material" as fuel. Starting was effected by a starter motor, thrust-regulation by changing the revolutions of the turbine pump. The combustion chamber was cooled with "M-material". In a power unit developed on parallel lines the combustion chamber was cooled with  $H_2O_2$ . This unit had a combustion chamber of cylindrical shape as compared with the almost spherical combustion chamber

of the 201—509 unit. Also the control system was rather different. This unit was not produced in series, yet it provided valuable experience.

The fighter aircraft Me 163, which was used in action, was fitted with the power unit 109—509. The same unit was incorporated in the interceptor-fighter "NATTER", designed by BACHEM and built by FIESELER, which took off vertically. The maximum velocity of climb of this aircraft was 10,000 m/min, and its ceiling was 15,000 m.

An improved type of this unit, the 109—509 A 3 unit, which for reasons of economy was fitted with an additional cruising jet of 400 kg thrust, was the motive power of the fighter aircraft Ju 263, which had a retractable carriage instead of the usual skids.

A slightly modified type of the same power unit was built, (though not in series) as an auxiliary propulsion device, into the tail-end of the jet-fighter Me 262. It gave this aircraft climbing properties which made it superior to all others.

The short time of operation of rocket-driven aircraft was a considerable handicap to their universal use. Therefore an attempt was made to develop a power unit which would be superior to the jet-propelled aircraft in climbing properties and speed, and superior to the rocket-driven aircraft as far as endurance was concerned. The experience gained since 1934 with ram-jet propulsion developments proved to be valuable in this respect.

Depending on the velocity of flight, air was pre-compressed in a diffuser and further compressed in a multi-stage axial compressor up to about 5 ata. This compressor was driven by an  $H_2O_2$  gas-steam turbine. The exhaust steam of the turbine and air were mixed in the combustion chamber; there they were burnt and discharged through the cooled reaction nozzle into the open atmosphere. The range of the Me 163 when equipped with this power unit would have been increased threefold. When the war ended, the first set operating on to this principle was on the test-stand. It would also have been particularly suitable for tele-guided missiles.

Research work on the ram-jet unit had been carried out, with occasional interruptions, since 1934. The experiments proved exceedingly successful, but a suitable test-plane of sufficient speed was lacking. For generating compressed air up to 3 ata, a compressor originally belonging to the gas turbine mentioned above was available. The efficiencies obtained agreed exactly with expectations. Ram-jet units, and units with an  $H_2O_2$ -nozzle attached to them, were tested

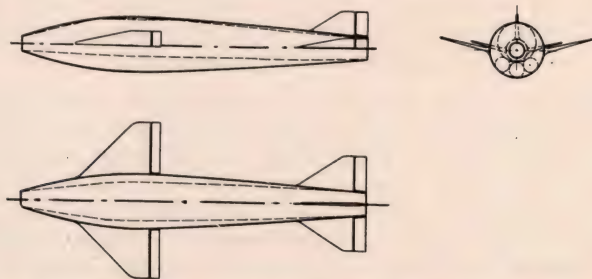


Fig. 15. Project of an aerial torpedo with ram-jet propulsion  
Total weight 1000 kg, payload 250 kg, velocity 300 m/sec, distance 300 km

with satisfactory results in high-speed wind-tunnels <sup>5</sup> producing an air-flow of up to 1000 kg/h.

This propulsion method was been proposed not only for propeller and helicopter power-units, but also for tele-guided missiles and robot-planes.

The project of an aerial torpedo with ram-jet propulsion shown in Fig. 15 dated from 1939. A similar project had already been submitted to the Army Ordnance Office in 1934. Based upon the experience with power units and catapults available in 1942, such missiles having a speed 40% higher than that of the V-1 might well have been constructed. Also a turbo-jet power unit with exhaust-reheat (afterburning) was suggested to the German Air Ministry in 1934.

Reference should now be made to the parallel development of a suitable  $H_2O_2$  power unit for torpedo propulsion. Here the requirements were particularly exacting: The starting time was hardly one second. The power unit was to be switched off and on again each time the torpedo, due to a failure of the depth-control apparatus, came above the water surface. The marine-torpedo was not to leave any bubble or oil traces. Finally, the unit had to withstand many months storage under tropical conditions. The possibility of fuel-leakages, which might have led to fuel agglomerations in the fuselage, always produced a certain feeling of uncertainty. Owing to the surface-decomposition still being imperfect in 1936 and due to the exceedingly short starting time required, the original method of working with pre-decomposition had to be abandoned. Instead hydracine hydrate-methanol was used for igniting. Finally, an attempt was made, with some success, to render the fuel (kerosine) self-igniting by means of nitrogen-free additives reactive to  $H_2O_2$ .

As far as air torpedoes were concerned, the bubble-trace problem was of minor importance, since the release of the torpedo will generally be observed. Thus, in this instance, the proved hydracine hydrate could be used.

In spite of the considerable technical difficulties which had to be overcome, the latest types of marine torpedoes had a high standard of safety and reliability.

As an example, take the "WAL", which was built in two types of 5 m and 7 m length and had a power unit of 500 HP. This set was equipped with fuel pumps, and therefore did not have delivery by compressed air. By this means, and also by injection of seawater, which made it possible to dispense with the transport of feedwater, the range of operation could be considerably increased as compared with previous types. The 5 m torpedo had a range of 8 km and the 7 m one a range of more than 20 km. The propellant gases contained 15% moisture, so that over-salting of the turbine was avoided.

Development had started with submarine propulsion. Almost all submarines were driven by turbine units operating with combustion. DIESEL-engines were thoroughly tested in addition. With a rather straightforward and reliable plant, which was tested during a thousand-hours trial run, after decomposition the steam-oxygen mixture was passed through a cooler or expansion turbine and then processed by the engine in the same way as air. The exhaust gases were cooled or condensed respectively by seawater and the  $CO_2$  was discharged overboard by a compressor. A so-called cyclic process was not necessary, since the combustion temperatures were like those produced with air operation.

In the case of the turbine cycle generally employed,  $\text{H}_2\text{O}_2$ , fuel and feed-water were delivered by metering devices into the decomposer or combustion chamber respectively, expanded in the turbine and were subsequently cooled or condensed in an injection condenser. The feedwater circulated and served to cool down the combustion gases from  $2200^\circ\text{C}$  to  $600^\circ\text{C}$ , the temperature which would do no harm to the turbine. Altogether, 5 test-stand units and 9 ship units were built. A large number of further units were under construction. Power outputs were 2000, 3000, 5000 and 8000 HP.

## 6. REVIEW

The numerous applications mentioned above showed the achievements obtained with  $\text{H}_2\text{O}_2$  within a few years. Furthermore, they demonstrate the confidence placed in  $\text{H}_2\text{O}_2$  by those charged with its exploration. This confidence was in the first instance due to the favourable physical properties of  $\text{H}_2\text{O}_2$  as compared with the two other important liquid rocket fuels, nitric acid and oxygen. These properties made it possible to build many thousands of rocket- and rocket-like power units with auxiliary propulsion sets and to handle them in active front-line service. In addition, the specific thrust is of the utmost importance<sup>6</sup>. Oxygen has the highest specific thrust per unit weight of fuel, though producing by far the highest temperatures. But for modern manned, or unmanned, missiles as well as for torpedoes and submarines the thrust per unit volume is the overriding factor, and here oxygen is hopelessly beaten.

On the other hand, 90%  $\text{H}_2\text{O}_2$  and 100% nitric acid have about the same specific thrust as regards unit of weight and unit of volume; maximum reaction temperature, however, being several hundred degrees higher in the case of nitric acid. This, due to the bubble line caused by exhaust gases, can obviously not be used for torpedoes or submarines. For manned aircraft, the choice should not be difficult for those who have gained experience with either fuel in practice. Nitric acid is, though, better suited for power units which, locked and sealed, will have to be stored for years on end. Therefore, experiments were started in 1942 with nitric acid.

The author wishes to express his thanks to all whose co-operation enabled him to describe here what once they had achieved, in particular to the experts of the former German Armed Forces, and to the workmen, mechanics, engineers and scientists of the former H. WALTER K.G., Kiel.

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## DISCUSSION

Dipl.-Ing. R. BRÉE (Wiese, Siegkreis, Germany): In his lecture, Prof. WALTER mentioned the BV 143, a teleguided missile developed by Messrs. BLOHM & VOSS (Dr.-Ing. RICHARD VOGT) and intended for use against ship-targets. This missile was similar to a torpedo, but noteworthy because the whole of its trajectory was to take place in the air. After release, the unit was to approach the water surface at a flat gliding angle, changing into a horizontal flight just above the water before actually touching the surface. The switching over from gliding into horizontal flight was to be effected by means of a mechanical feeler, and Prof. WALTER was of the opinion that relevant tests had been successful. I am sorry to say that this was not so. One would have been extremely happy if this problem could have been realized. The automatic change-over into a horizontal flight would have reduced the teleguidance to a mere side-correction task. Unfortunately, this was asking too much, since the short pullout available with the use of mechanical feelers was by no means sufficient. A height of only about 2 m was involved! Not only were very high accelerations required, but also a change from (driven) gliding flight to horizontal flight without any adjustment, i.e. a change which could not be realised in practice. Since the mechanical altimeter proved useless, one might have envisaged the use of electrical altimeters. These, however, were not available in the form required for this purpose at that time, so that further development of the BV 143 was discontinued.

# THE DEVELOPMENT OF THE V-2 ROCKET ENGINE

MARTIN SCHILLING \*

It is my privilege to lead us, in a brief discussion, through the major phases of the German liquid oxygen—alcohol rocket powerplant development which culminated in the prototype and production model of the A-4 engine, the most powerfull rocket powerplant of its time and which has been surpassed only in post-war years.

## 1. THE CHOICE OF PROPELLANTS

Before we turn our attention to the technical subject proper, I would like to make a few remarks about the history of the development of liquid rocket propulsion systems. In this discussion today, we will deal only with developments undertaken for the Army Ordnance Corps (HEERESWAFFENAMT). The Army pursued the development of true rockets with rocket power plants only, the latter ones on liquid oxygen—alcohol and nitric acid—fuel basis. I will restrict my remarks to the former system, as Mr. VON ZBOROWSKI, in the following paper, will adequately cover the field of acid—fuel rocket motors. When this Army Ordnance project was established at Kummersdorf in the early thirties under the then Capt. DORNBERGER and Dr. VON BRAUN essential, although modest results of rocket powerplant component developments were already available. Collectively, these contributed to the Ordnance Corps resolve to incorporate liquid-propelled rockets in the weapon system development. It is a pleasure to mention, at this time: Prof. OBERTH; the Rocket Experiment Station (RAKETENFLUGPLATZ) Berlin with Messrs. NEBEL, RIEDEL, VON BRAUN, and HUETER; MAX VALIER and co-workers; and Dr. HEYLANDT and the contributions by his staff. Closer scrutiny undoubtedly would add to this distinguished list. With modest means, and without federal aid, these scientists and engineers pioneered in a new field which only later moved into the foreground of profound general interest. Furthermore, I would like to bring to your attention Dr. GODDARD and his influence on the rocket engine development. As a single individual, he made important contributions and gave a distinct note to the new art of rocketry. The German development efforts did not profit from his prolific work, and were actually carried out in almost complete ignorance of his achievements. This was the result not only of Dr. GODDARD's strong inclination to secrecy, which is reflected in his patent applications, but

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also of the official disruption by Germany of the exchange of information in this field on national and international levels. This is understandable when the early participation by the Ordnance Corps and its material support of rocket development is taken into account. So the field of rocketry, as many others, may well have experienced the duplication of efforts and development results which are bound to come about through the application of the same sound basic theories and development principles.

The choice of the liquid oxygen—alcohol combination is by now historical. Liquid oxygen as the purest form of any oxydizer promised to present little difficulty in atomization, ignition, and control of a high-pressure combustion process in a rocket motor. Besides, it was well known from other industrial applications. Alcohol, both ethyl and methyl, were also available in large quantities, were reasonably priced, and delivered according to satisfactory commercial specifications. The combustion temperature in combination with liquid oxygen is agreeably low, but a reasonable combustion efficiency is nevertheless maintained because of the low molecular weight of the exhaust gases. This fact was considered to be important at that time when practical experience was still low and the silent threat of momentous heat transfer problems was ever present in the design offices and outside in the modest test pits. Regenerative cooling had been introduced as an essential feature of liquid fuel rocket motor design, and here, also, alcohol proved to be a superior coolant compared with other hydrocarbons. It was further established early by computation and experiment that the admixture of water resulting in 75 % by weight alcohol content resulted only in a modest reduction of motor efficiency, but added greatly to the safety margin of structural design for the reasons outlined above.

There, in short then, are the basic reasons for the choice of this propellant combination: liquid oxygen — 75 % alcohol. For the purposes of systematical research, the whole practical range of oxydizer-fuel ratios was investigated,



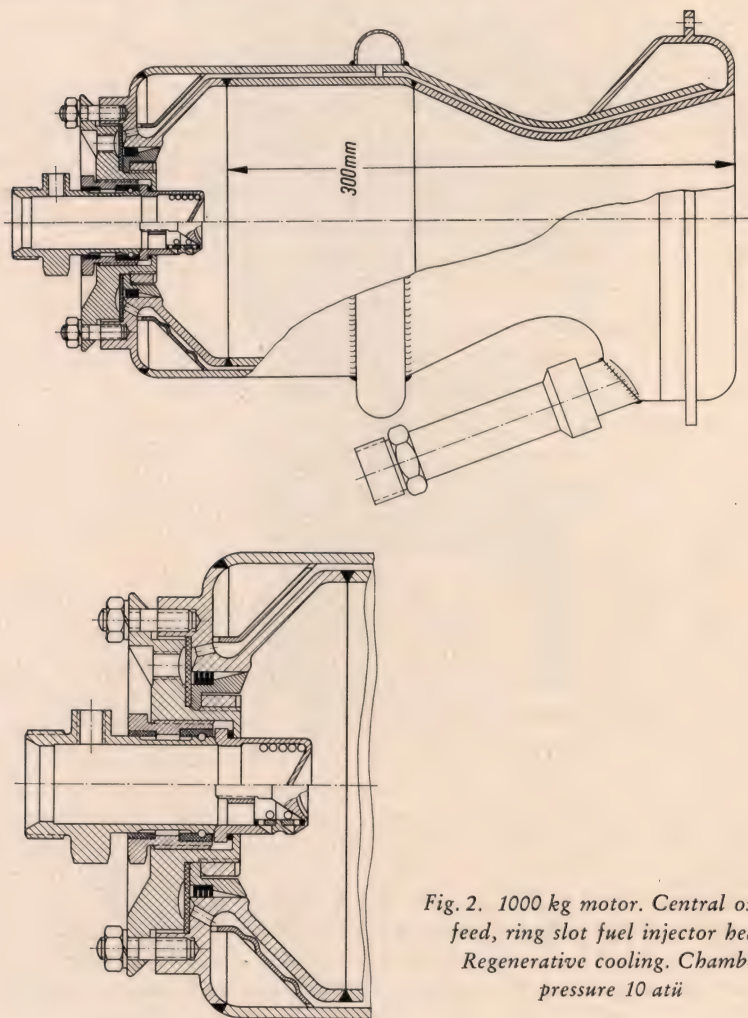
*Fig. 1. Example of completed liquid oxygen-alcohol motor development: Twin Jato power-plants, 1000 kg thrust each; acceptance test runs for Air Force, Peenemünde 1940*

both fuel rich and fuel lean. Actual operation proved to be smoothest in the fuel-rich region; this was accepted as a fact for the V-2 motor.

## 2. EARLY DETERMINATIONS

The rocket motor development began in earnest and with spectacular success in the late thirties, first under Dr. WERNHER VON BRAUN, and later under Dr. WALTER THIEL who led the V-2 propulsion system development nearly to completion before he was killed in the air raid on Peenemünde on 3 August 1943.

As so often in life, there is a little story and moral connected with this motor development: In 1938, tests were run pretty reliably for a duration of



*Fig. 2. 1000 kg motor. Central oxygen feed, ring slot fuel injector head. Regenerative cooling. Chamber pressure 10 atü*

approximately 60 sec each, which was considered necessary for a large liquid propelled rocket, with exhaust velocities around 1800 m/sec or roughly 75 % of the theoretically feasible value. The basic unit was a motor of 1500 kg thrust at approximately 10 atm combustion pressure. From this it was concluded that 2000 m/sec exhaust velocity, or, as we prefer to say today, a specific impulse of 204 lb sec/lb, was feasible and could be achieved under accelerated development schedule. This was one of the major predictions which triggered the development of the V-2 and became a basic element of the preliminary design of the missile. Needless to say that this goal was reached only by taking great pains and after a prolonged see-saw battle between success and failure. Further, it was also the object of long and heated arguments. Finally, when, after a very careful error analysis we were in a position to bring statistical evidence to show that the design goal had been surpassed by 3 % in efficiency, with a bonus of about 15 km range increase above the guaranteed range, the feelings of uncertainty and misgivings gave way only gradually to one of deep satisfaction and pride. Thus, a major achievement in the field of rocketry entered the scene very modestly and in a very human way.

The step from a 1.5 ton unit to a 25 ton thrust unit is keen, to put it mildly. It was therefore decided to do an intermediate step first. We learned with this one that there is no easy approach to an intermediate step, but that each new development is a full-scale problem. Soon therefore, primarily for time reasons, this intermediate 6-ton motor development was abandoned and the big problem of a 25-ton thrust unit was tackled directly.

### 3. ROCKET MOTOR DESIGN FEATURES

At this time, a few remarks about motor design are in order.

Besides the mysteries of the combustion process which he simply believed in and entrusted to the Chemo-Physicist, the design engineer's main guidance was STODOLA's treatise on steam turbines. By continuous monitoring and cross-checks with missile preliminary design studies, we gradually learned to avoid certain pitfalls of too sophisticated approaches, for example in the LAVAL nozzle designs. Two of the criteria so established were:

1. Exhaust gas flow is sufficiently inhomogeneous to make supersonic expansion in a straight cone with normal sheet metal surface finish quite adequate.
2. A short cone with moderate expansion ratio ( $15^\circ$  cone angle) adjusted to approximately 0.8 atm exit pressure results in optimum missile performance considering motor and missile tail section weight, thrust coefficient and total impulse, base drag, and jet vane control performance.

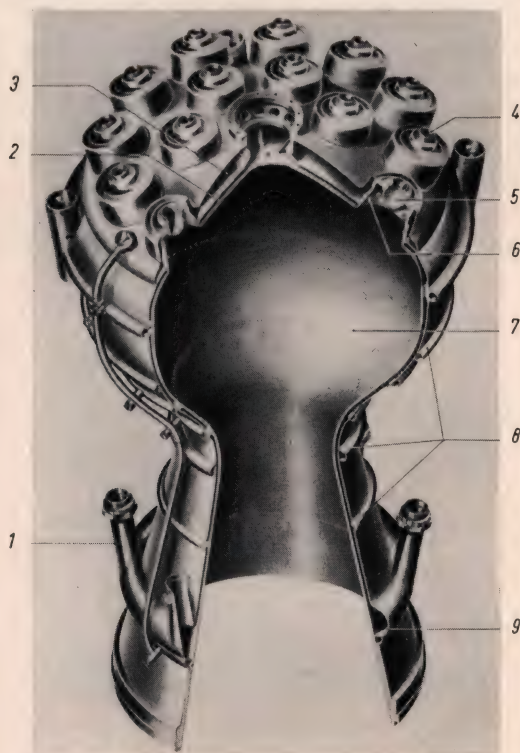
This did away with several awkward experimental motor models of large size and weight and concentrated the effort on the type of engine you eventually saw incorporated in the A-4. A lot of pioneer work in this area was done by the Institute of Prof. WEWERKA at Stuttgart. Others to be mentioned at this time for essential contributions are, Prof. HASE of Hannover, and Prof. VIEWEG of Darmstadt, who contributed largely to the field of powerplant instrumentation; Prof. SCHILLER of Leipzig, for investigations in the field of regenerative cooling; Prof. CARL WAGNER of Darmstadt, for the mathematical solution to the combustion process; Prof. PAUER and Prof. BECK of the T.H. Dresden for

clarification of atomization processes and the experimental investigation of exhaust gases and combustion efficiency, respectively.

On the Peenemünde staff, it was predominantly Dr. WALTER THIEL, as whose deputy I had the honour to serve until his sudden death, with a dedicated staff, among them in the design RIEDEL III, DANNENBERG, and LINDENBERG; in the development RIEDEL II, HELLER, EHRLICHE, ZOIKE, ZANGL, PILZ, MÜNZ, and BEDÜRFTIG, who can take the credit for the establishment of the first large and efficient power plant of this kind.

But now back to the interesting motor problems:

The fight to avoid hot spots and for control of the huge heat flow rates through the motor walls was ever present. Measurement on experimental A-4 motors had established in 1940/41 heat flow densities of the order of



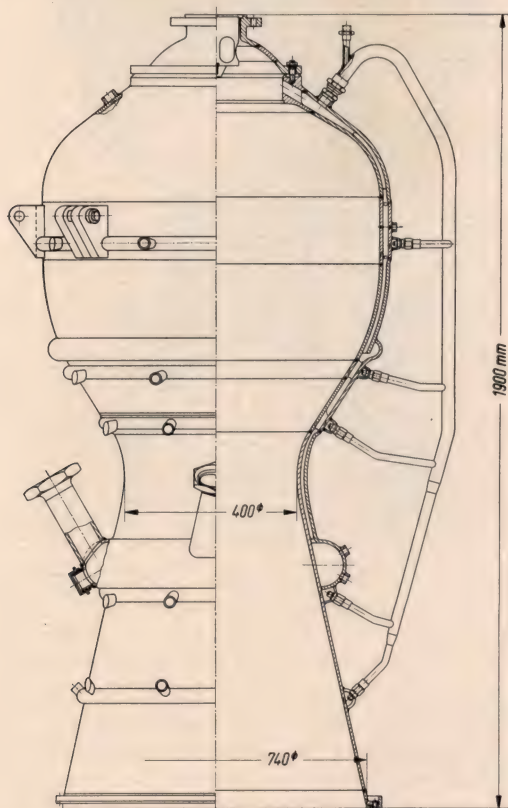
*Fig. 3. Cut-away view of A-4 motor production model. 25,000 kg thrust; chamber pressure 14.2 atü; specific impulse 210 sec; weight approximately 400 kg*

- |                                 |  |
|---------------------------------|--|
| 1 = Fuel intake                 | 2 = Lower head chamber<br>(individual head element with<br>oxygen feeder line connector) |
| 3 = Upper head chamber          | 4 = Nipple for oxygen feed line  |
| 5 = Oxygen shower head injector | 6 = Fuel injection nozzles   |
| 7 = Combustion chamber          | 8 = Expansion beads  |
| 9 = Fuel intake bead            |  |

*Notice further the three rows of holes in combustion chamber and expansion nozzle for admission of alcohol for internal film cooling*

3,000,000 kcal/m<sup>2</sup>h °C. This problem threatened to limit the motor efficiency although regenerative cooling was fully understood and employed until the breakthrough came through internal fuel film cooling. The V-2 motor shows the cruder of two schemes: 10 % of fuel flow used for film cooling reduced the heat transfer by approximately 70 %. Mr. POEHLMANN refined this by oozing, not injecting the film fluid into the chamber, and achieved the same beneficial effect with a considerably lesser amount of fuel. For all practical purposes, this method solved the problem.

We now enter the discussion of another intricate problem, that of fuel injection, atomization, and combustion. Throughout all the early years up to 1942, you see in all sorts of variations the arrangement of a central oxygen feed and spray nozzle configuration, with alcohol ascending to the motor head through the cooling jacket and entering the combustion chamber through a large number of individual injection nozzles having various characteristics. The largest number of test runs were made with 1000 kg to 1500 kg motors, and it is no accident that this type of injector head found its application in the V-2 motor proper. Admittedly, this was a monstrosity and a plumber's nightmare with its many feeder lines, but was the only design ready for production under accelerated delivery schedules in 1943.



*Fig. 4. A-4 25,000 kg rocket motor with one central injector plate*

We obtained at the same time the clarification on advantages and disadvantages of another major design principle on which much effort had been spent — that of the pre-combustion chamber. This two-stage combustion process is beneficial as long as fuel mixing and atomization are imperfect: the combustion chamber volume is increased, turbulence in the exhaust gas likewise, unfortunately accompanied inevitably by a non-isotropic pressure drop and therefore a loss in efficiency. This was still acceptable for the relatively efficient V-2 motor, the more naturally for the more primitive designs of prior years.

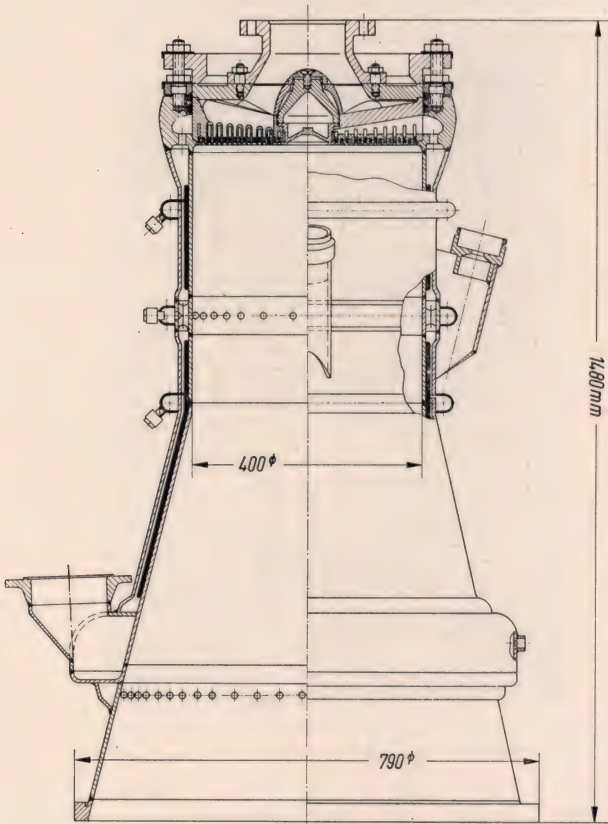


*Fig. 5. Experimental motor of version shown in Fig. 4. Motor cooling by internal film cooling only*

A whole variety of motors had been built: horizontal injection in star configuration, ring-shaped precombustors, as e.g., in the so-called POEHLMANN motor, and others. What finally remained, was the relatively modest inclination toward this principle in the V-2 engine.

Starting in 1942, a more vigorous and successful attack was made on the injector problem by introducing in experimental units the injector plate as it is standard in today's more advanced rocket motors. The advantages were manifold: first, the multitude of fuel and oxidizer feeder lines disappeared thus producing a cleaner design of improved reliability. Next, a controlled

association of fuel and oxidizer particles was enforced which was to result in better control of combustion parameters and was to assure improved combustion efficiencies. Furthermore, simple fuel line connections are a pre-requisite for gimballed suspension of rocket motors for in-flight missile control, which feature was under active consideration. The experiments which were carried out during the remaining war years in motors up to and including the size of V-2 engines, proved to be fully effective at the price of introducing in its full severity another phenomenon practically unknown in the earlier designs: dynamic instabilities in form of chugging and screeching. They were greatly reduced, but not fully overcome when our development effort came to a stop late in 1944 because of the imminent end of the war.



*Fig. 6. Tubular design of A-4 rocket engine*

Into this same time period falls another more systematic investigation: that of what we call today the motor characteristic length  $L^*$ . The V-2 combustion chamber volume was considerably decreased through these efforts, but an extreme, a tubular combustion chamber of throat area diameter over its entire length proved to be too radical and was abandoned after some early trial runs.

#### 4. FEEDER SYSTEMS

Extensive preliminary design and feasibility studies established economical brackets for gas pressurization and pump fed systems. The V-2 naturally fell clearly into the latter category. Higher pump feed and combustion chamber pressures up to 40 atü would have resulted in increased performance, but time pressure did not allow us to move too drastically away from the wealth of existing experimental data which were grouped around 10 atm chamber pressure. The V-2 motor, therefore, is a conservative compromise.

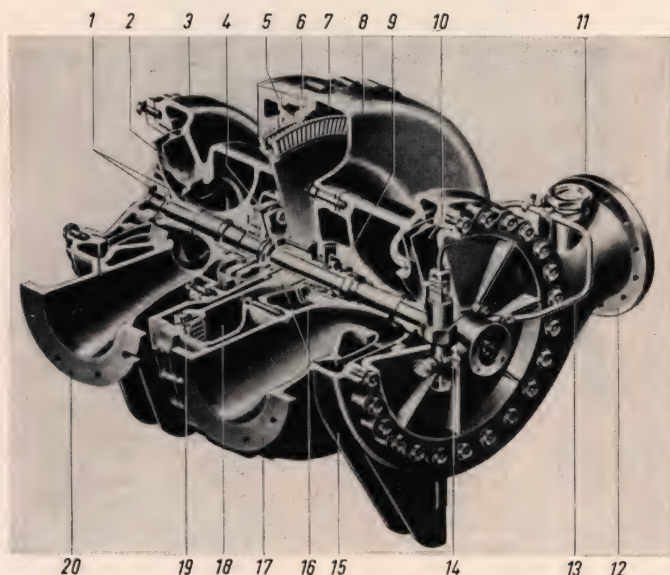


Fig. 7. Cut-away view of turbo-pump assembly

- |   |                                     |
|---|-------------------------------------|
| 1 = Oxygen pump bearings                      | 2 = Oxygen pump impeller            |
| 3 = Oxygen pump                               | 4 = Coupling                        |
| 5 = Nozzle case                               | 6 = Live steam casing               |
| 7 = Waste steam casing                        | 8 = Steam turbine                   |
| 9 = Alcohol pump bearing                      | 10 = Alcohol pump impeller          |
| 11 = Pressure nipple (for pressure regulator) | 12 = Alcohol pump pressure nipple   |
| 13 = Alcohol leakage line                     | 14 = Safety mechanism (spring lock) |
| 15 = Alcohol pump                             | 16 = Turbine shaft seal             |
| 17 = Alcohol intake                           | 18 = Turbine wheel                  |
| 19 = Turbine blades                           | 20 = Oxygen intake                  |

Various pump systems were investigated for application to missiles. It finally resolved itself into a good compromise design established by KLEIN, SCHANZLIN and ODESSE: two centrifugal pumps of approximately 4300 rpm, on the same shaft with a central two-stage steam turbine of approximately 500 HP. Pump suction heads around 1.5 atm, delivery pressure around 20 atm, pump and turbine housing: light metal alloy. A few tough problems had to be overcome:

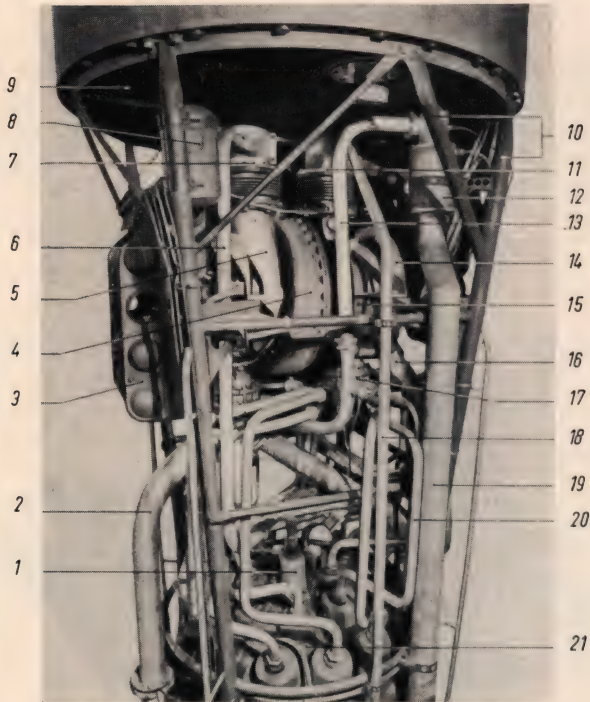


Fig. 8. Propulsion Unit Assembly

- |  |   |
|--|---|
| 1 = Main alcohol valve                       | 2 = Alcohol feed line to combustion chamber   |
| 3 = Compressed air battery                   | 4 = Turbine                                   |
| 5 = Alcohol pump                             | 6 = Alcohol circulating line                  |
| 7 = Alcohol intake                           | 8 = Colour reservoir (for test missiles only) |
| 9 = Flow bulkhead                            | 10 = Thrust frame                             |
| 11 = Oxygen pump inlet                       | 12 = Oxygen tank vent valve                   |
| 13 = Pressure-regulator pipe for oxygen tank | 14 = Live steam pipe                          |
| 15 = Oxygen pump                             | 16 = Live steam line for 14                   |
| 17 = Main oxygen valve                       | 18 = Oxygen topping line                      |
| 19 = Oxygen vent line                        | 20 = Oxygen feed lines                        |
| 21 = Combustion chamber head                 |   |

warping of housing because of high temperature difference between steam (425 °C) and liquid oxygen (— 183 °C); lubrication of bearings in alcohol and oxygen pumps; seals and gaskets and choice of alloys, to mention only a few.

The advantages of a high-speed steam turbine and a reduction gear between turbine and pumps were recognized but did not materialize because of time reasons. An improved pump design by SCHUPP was available in 1943, but was not released for mass production.

Now the question of the turbine steam supply will be on your minds. Here eventually, we borrowed heavily from WALTER, Kiel, using hydrogen peroxide decomposition. A first attempt to adapt and improve a torpedo steam generator failed because of numerous details (valves, combustion control) and was

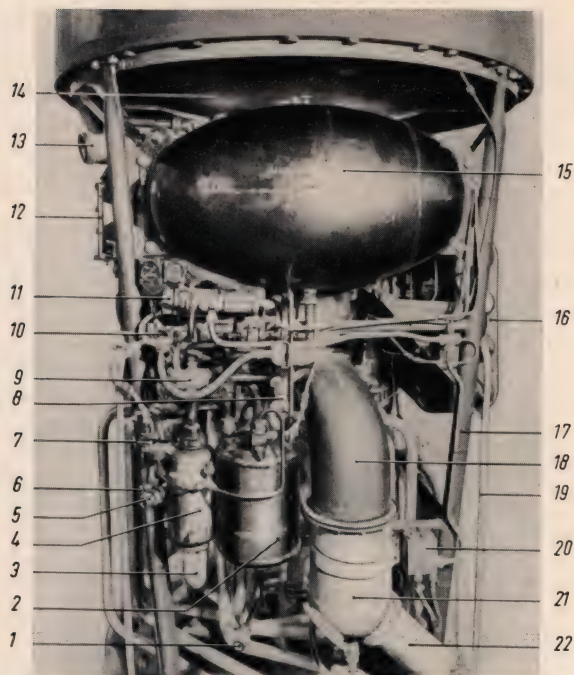


Fig. 9. Steam Generator Assembly

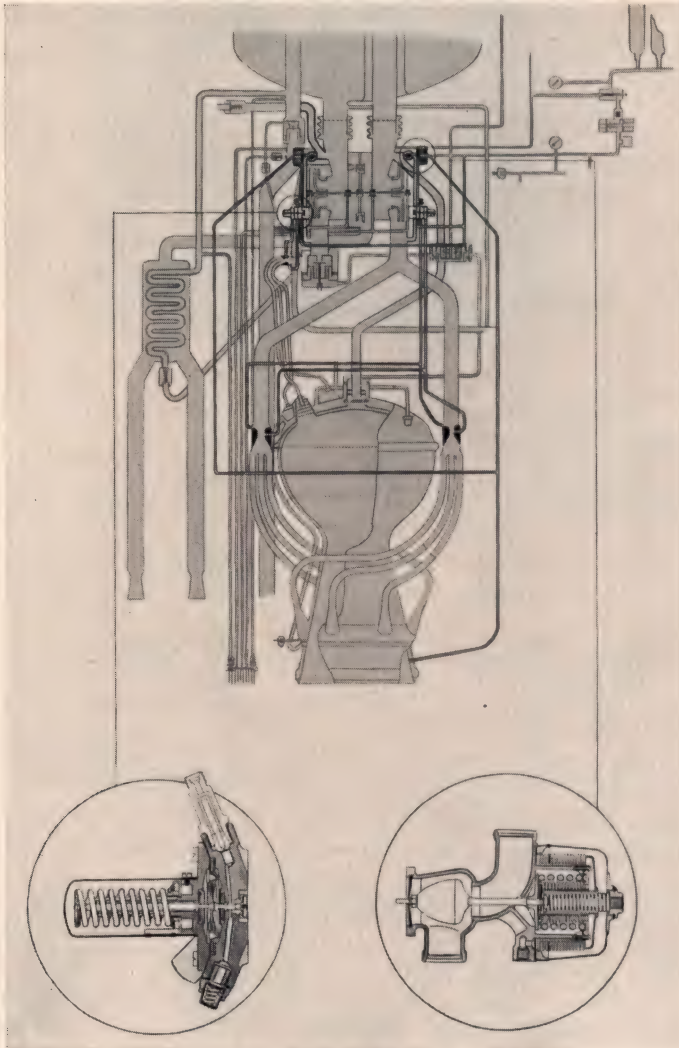
- |  |   |
|--|---|
| 1 = Flushing valve for liquid catalyst container   | 2 = Liquid catalyst container                           |
| 3 = Live steam line                                | 4 = Decomposer  |
| 5 = Flushing valve for hydrogen peroxide container | 6 = Vent line for liquid catalyst container             |
| 7 = Catalyst pressure switch                       | 8 = Control valve for 25-ton valve                      |
| 9 = 25-ton valve                                   | 10 = Main valve for hydrogen peroxide                   |
| 11 = Air pressure reducer                          | 12 = Intermediate distribution box                      |
| 13 = Oxygen filler valve                           | 14 = Flow bulkhead                                      |
| 15 = Hydrogen peroxide container                   | 16 = Compressed air battery                             |
| 17 = Thrust frame                                  | 18 = Exhaust elbow                                      |
| 19 = Fender for covering the tail                  | 20 = Control battery for oxygen and alcohol main valves |
| 21 = Heat exchanger                                | 22 = Steam exhaust line                                 |

shelved. The effort then concentrated on a chemical steam generator as mentioned before.

A pump fed version of this system promised a significant reduction in weight, but was discontinued because of excessive difficulties with valves and regulators. A decomposer using solid catalyst — silver screen or porous material coated with a cobalt salt — was available in 1944 and worked satisfactorily. The missile specification prescribed 72% by weight concentration of  $H_2O_2$ ; it was finally commercially available in concentrations around 80%. This provided for an adequate safety margin for losses as a result of transfer and storage.

In concluding on turbine steam supplies I would like to mention that 1944 saw the successful completion of a component development in the form of a liquid oxygen—alcohol steam generator, but no missile application was any longer possible.

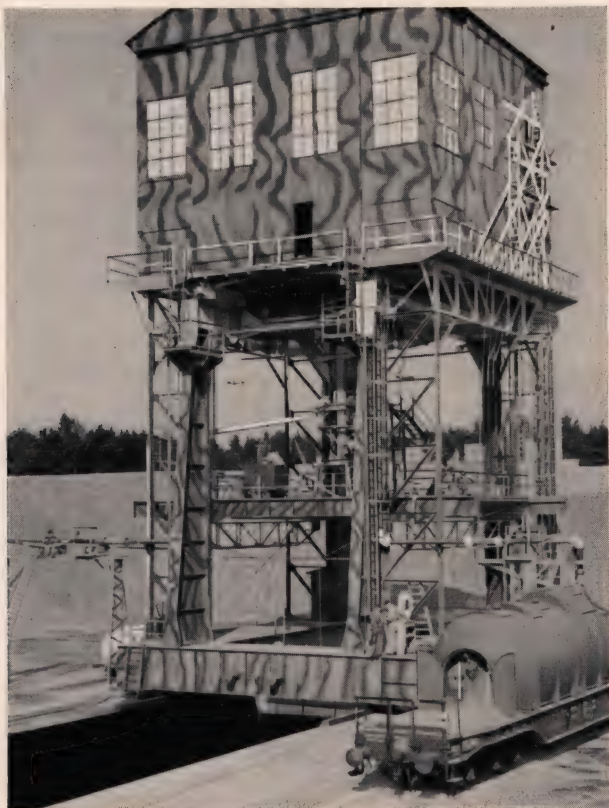
A few concluding remarks are in order on power plant regulation and controls. A ballistic missile of the V-2 type is intended to fly an optimum standard ascending trajectory under powered flight. Furthermore, flight performance with respect to range is adversely affected by the amount of residual propellants still in the containers at the termination of propelled flight. This suggests both the development of a thrust regulator and of propellant mixing ratio control. Both developments were actually undertaken and nearly



*Fig. 10. Mixing ratio control, schematic*

completed at the end of the war. Accuracies achieved were of the order of 1% and below.

Time does not permit to mention in sufficient detail the vast amount of test facilities designed, built, and operated for propulsion development. The last Figure will convey the impression of the size and complexity of a propulsion system test stand.



*Fig. 11. Propulsion system test stand. A-4 powerplant installed.  
Layout of stand for 100 to unit*

Retrospectively I can say that, based on a major breakthrough in small motor development around 1938, it was possible to scale up in one step by a factor of roughly 18 and to create a modern pump-fed rocket power plant of acceptable efficiency in a three year period; ready for production after one more years effort. Understanding, generous support and leadership by the Ordnance Corps represented by General DORNBERGER, adequate funds, effective industrial and academic support, superior achievements by the university staffs, a competent and enthusiastic team at Peenemünde, and good luck — all of these contributed to successful completion of this major development.

## DISCUSSION

Mr. A. R. WEYL (Dunstable, Beds.): Our subject is the history of guided missiles in Germany; my observations have, hence, bearing upon the matter under discussion. In 1923, a schoolmaster of a Siebenbürgen grammar school published a booklet dealing with the design of a high-altitude rocket. The proposals given in that booklet show a most remarkable similarity, in many design features, apart from the fuel pumps, with the rocket engine of the V-2. Has this publication by HERMANN OBERTH been known at the Peenemünde establishment? If this has been so, is it not puzzling that neither the lecturer nor anybody connected with the Peenemünder effort have cared to recall OBERTH's pioneering work? For anyone from the outside, it is a quite amazing that an inspired research worker of his calibre who was widely known in this field, has been kept out of the realization of his ideas?

Dr. SCHILLING: It goes without question that rocketry owes much to Prof. OBERTH. He is a great personality and has given leadership and inspiration. Most of these things are too well known. I can speak authoritatively on this subject, as I have had the privilege to have been associated with him for several years in Peenemünde during the war. In closing, I would like to mention that Prof. OBERTH is now with us again in the United States.

Admiral FAHRNEY (Philadelphia): Since I was intimately associated with guided-missile development in America from 1936 to 1950 as a project officer, and finally as the director of the Guided Missile Division for the Bureau of Aeronautics, I have had a lot of experience with the direction of these programs. As we all know, no accomplishments on a scale as vast as the V-2 program can be realized unless the engineers and scientists are supported, and unless there is a favourable climate of official approval under which to work. This climate is provided only by government activity or a large industry with ample funds. In my experience it is extremely difficult to sell a program for new weapon developments to the authorities because of serious competition with old programs for funds and for people. Since, for that period of missile development the V-2 and Peenemünde were such gigantic projects, it would be most helpful to me and, I am sure, to other military personnel present if General DORNBERGER would elaborate on the methods which he used to initiate and to keep moving complex programs of this character.

General Dr.-Ing. E. h. DORNBERGER (Buffalo): In answering this question, please don't expect me to give you a panacea which may be used today. Times have changed.

First of all you have to take into consideration the environmental conditions prevailing in Germany at the start of the development of the V-2 and of the later guided missiles. In the late twenties, the German Army was qualitatively and quantitatively restricted in its armament by the Treaty of Versailles; therefore, much effort of the Army Board of Ordnance (HEERESWAFFENAMT) was directed towards loopholes in this treaty. Rockets seemed to offer an opportunity of over-coming part of its restrictions. A couple of years later we were in a state of re-armament and everything became much easier.

The Army Board of Ordnance was very fortunate to have, at that time, as Chief of the Ballistics and Ammunition Division of its Development Depart-

ment, a man of outstanding calibre, the Col., later General, Prof. Dr. KARL BECKER, who soon became Chief of the Development Department and finally Chief of the Board of Ordnance until his death in the Spring of 1940. Under his successor, General EMIL LEEB, the situation did not change appreciably. General BECKER was an excellent Ballistics and Ammunition expert with a clear vision into the future as far as armament was concerned. Certain funds were available to him to use at his discretion to further research and development of modern classified weapons. He did not have to account for this money. General BECKER, under his own responsibility, and with more or less tacit consent of his superiors, initiated research and development in the rocket field. After the initial success with powder rockets, he was anxious to get on with liquid fuel rockets. Since both developments were done in military installations and under the leadership of younger officers with engineering backgrounds, rockets soon became a "pet child" of the Board of Ordnance.

During the development phase of the first solid fuel rocket weapons, as during the V-2 development program, there were no military requirements ever written for these new weapons by higher level offices or the General Staff. The Board of Ordnance never asked for them. At that time where everything was still very hazy and unproven in the rocket field, military requirements could best be established by technically trained officers being in the closest contact with rapidly changing technical progress. Therefore, we, in the Board of Ordnance, had to make up our own minds as to what a useful weapon might look like. For example, the Department of the Artillery refused for years to bring solid fuel rockets as weapons into operational use and saw no need for any long range liquid fuel rocket. Not before the Department for Gas Warfare was established (in 1934) did solid fuel rockets become an officially recognized weapon. What I want to say with this is that in the first years we, in the Board of Ordnance, did our development work in the rocket field independently and not hampered by directives from higher echelons.

Later, when we had to ask for more funds, we had to have the consent of certain high-ranking Army Chiefs. Fortunately there were a number of Generals in key positions who looked favourably at this now growing-up Army child, and helped with funds as far as they were able to without making too much noise about it, and with no fear of being investigated by any governmental institution. On account of its "top secret" character, the expenditure of these funds was under the exclusive control of the Development Division Chief (WA PRÜF 11) and was not subject to any normal Army Comptroller's office. To mention only a few of these visionary Generals, whom it was not difficult to convince: the Chief of the German Army High Command, Col. General VON FRITSCH, and his successor, Fieldmarshall VON BRAUCHITSCH, the one who (during the war) became the Chief of the Home Army, Col. General FROMM, and the Chief of the Inspection Department (ALLGEMEINES HEERESAMT) General OLBRICHT, and his successor, General KEINER. Of tremendous importance in our effort to win them to our side was the fact that we convinced them with facts obtained from the accomplishments we had already achieved. And, as it was in the German Army, when a superior had made up his mind, no subordinate would have dared to disagree.

At that time there were no military offices at higher level who had any experts in the rocket field. The Chiefs had to accept our word. There were

no experts in Technical Institutes or Governmental Institutions who could evaluate our program or make any recommendations. The responsibility of, and the authority for, the entire program from research to going in action always remained with one single Board of Ordnance Division (WA PRÜF 11) which at the end of 1943 was split up into two development divisions (one for solid fuel rockets, one for liquid fuel rockets), and the Long Range Weapons Special Commissioner of the Army (Beauftragter zur besonderen Verwendung des Heeres, BzbV Heer) who reported directly to the Chief of the Home Army. Thus, by avoiding a variety of offices at different military levels, any possible necessary decision was made by discussions among the leading Army Generals who, at that time, were quite willing to take many risks to further the program.

With only one single division working on this project and having the facilities (Peenemünde and Experimental Station West at Kummersdorf) as sections or groups of this division reporting to its Chief, our military superiors were only asked questions on very important matters. Hardly any reports asking for a decision were sent up the ladder of Command in the Army. It was not necessary. Once a year a presentation of the progress was made to a selected group of high echelon Generals, but without asking for any decisions. Most of the high ranking officers in the German Army up to the Fall of 1944 did not even know that liquid fuel rockets had been developed. With the highest Army Generals supporting the program and having confidence in its management, everybody concerned within the Army did his best to help. Everything would have run smoothly if Party Organizations and the German Government (HITLER and the Ministry of Ammunition) had not been needed in order to start mass production and to go into operation.

The biggest task of the program ever undertaken during the war was to convince HITLER. He was first familiarized with the liquid fuel rocket program in 1934. Interested in all branches of Technology, he admired the goal at which we were aiming, but only from the viewpoint that the Germans were the first to develop such large rockets. He saw, however, no military use for them. His mind was directed toward guns, tanks, and aeroplanes. Therefore, he dropped us from the priority list at the beginning of World War II. Even with all our presentations, reports, brochures, and analyses, we could not change his mind. He allotted us just enough money to continue on a very small scale. We could only continue with the support of the Chief of the High Command of the German Army. When the military situation deteriorated in 1941, we bombarded HITLER's Headquarters with all kinds of material in rapid sequence, and made use of the readily offered help of the Secretary for Armament (Prof. ALBERT SPEER) with presentations; but these were in vain. HITLER dreamed that the big rocket could never reach England. At this time SPEER initiated an investigation of our work by industrial experts. The result was favourable. Simultaneously we had our first successful firing. Nevertheless, many more presentations with films and data, and a further deterioration in the military situation were necessary to ultimately compel HITLER to give us top priority.

Here you have in a nutshell the battle for acknowledgement of modern military rocketry. High ranking officers with vision into the future fought shoulder to shoulder with the Development Division of the Board of Ordnance to smooth the way for a revolutionary new weapon system.

# BMW-DEVELOPMENTS

HELMUT PH. G. A. R. VON ZBOROWSKI \*

## 1. INTRODUCTORY

The BAYRISCHE MOTOREN-WERKE (BMW) charged the author in autumn 1939 with the organisation and initiation of a rocket-engine development, and he directed this till the end of the war. The following discussion is based upon the work done during that period.

Part of this development has already been described by the author during a session of the DEUTSCHE AKADEMIE FÜR LUFTFAHRTFORSCHUNG held on 5th August 1943<sup>1,2</sup>; the subsequent work has been repeatedly published in technical journals after 1945<sup>3-14</sup>. It is, therefore, not deemed necessary to repeat an account of this descriptive information. Instead, the author's intention is to discuss the principles which guided the work, and to show, with the help of various examples, which results were obtained by applying these principles. Since these are of permanent and more universal value, they should prove of interest for engineering in general.

The discussion must be restricted to work done in Germany till 1945. Unfortunately, this gives little scope to do full justice to the progress actually achieved, as far as the contribution by BMW is concerned: more recent and most important developments in the field of guided-missile and aircraft design form logical steps based upon the war-time efforts of BMW.

This work was supervised by flight staff engineer HEDWIG from the office of general engineer EISENLOHR in the RLM.

The author wants to take this opportunity of expressing his gratitude to all his assistants, especially to the test stand mechanics, foremen and test stand engineers for their unselfish and devoted collaboration. Of all the numerous and splendid assistants at the end the following should receive special mention: Flight construction engineer ZUMPE as first assistant, the supervising engineers SCHELL and SCHNEIDER as directors of the test plants Zühlsdorf and Allach, Dipl.-Ing. DREYER and Dr. RISTAU as directors of the rocket development groups, and Prof. Dr. techn. GRUBITSCH as assistant for chemical ignition.

## 2. GENERAL PRINCIPLES GUIDING THE DEVELOPMENT

### 2.1. Quality Parameter

The development engineer as well as the industry itself found themselves again and again confronted by engineering problems which were subject to the most rapid changes in the official appraisal. It happened repeatedly that just

\* Dipl.-Ing. — Formerly: Chief-Manager of BMW Development Plants, Allach, Bruckmühl and Zühlsdorf. — At present: General Manager of Société Anonyme Bureau Technique Zborowski (BTZ), Brunoy, S. & O., France.

before a hotly demanded development could be brought to a conclusion, and often enough, after it had been achieved that, by a "snap" decision of the powers-that-were, the result was no longer wanted. Sometimes even when production had already been prepared or just begun to flow, orders were forthcoming to stop all the work and to attend to something else.

On the other hand, apart from such vacillation of the official mind, a multiplicity of often much variegated possibilities of solution presented itself for one and the same clearly defined problem, and this demanded a wise selection.

It became painfully obvious that neither the authorities, nor the industrial technicians had a clear, simple and universally applicable scale to use for the purpose of such selection.

Since 1939, i. e. from the beginning of the development considered here (then under the supervision of Director Dipl.-Ing. HARALD WOLFF), BMW applied a "quality parameter" for guidance; this expresses the all-round economy and efficiency of a weapons system. It can easily be shown that peaceful engineering as well as the armament technique are subject to one and the same dominating factor, and that is the over-all economy.

From 1939, BMW thus employed tables of specific power-plant costs to define the economy of powerplants in a way which permitted a fair comparison. These costs were uniformly referred to the free impulse. When, at a later stage, BMW took up preliminary projects of entire guided missiles and aircraft, the selection was again based upon the specific effort considering the complete craft in relation to the operational function for which it was designed.

In weapons production, the basic effort consists not only of materials and working hours but also of expense in human lives; hence we had to prefer a quality parameter to a mere evaluation of production costs (comparable to the more recent charming American formulation of "Kill-Costs").

This led to an exclusive and paramount evaluation scale which held for any item of equipment: Quality Parameter = "Effect obtainable" divided by "Effort required by the nation's capacity".

To obtain a non-dimensional figure, the effect of the item must, of course, be also expressed in units of the nation's capacity.

## 2.2. The Most Practical Propulsion Plant

The quality parameter is substantially affected by the selection of the most practical powerplant in any machinery but in particular, for weapons; another factor is the choice of the most suitable method<sup>15</sup>.

The term "powerplant" should actually be interpreted in the broadest sense since these considerations apply quite generally. Bearing in mind, however, the limitations to which this Seminar is subject, the selection of powerplants is restricted to propulsive devices in the narrow sense of the pre-1945 period such as: Solid- and liquid-propellant rocket-engines; ramjets; resonant pulse-jets; turbojets; propeller turbines; reciprocating piston engines with spark- and with compression-ignition. A first step for the solution of a weapon-engineering problem is, consequently, to discover which is the most practical kind and type of propulsion.

Considering the economy of an aircraft, the influence of the specific impulse costs is much less than that of the specific impulse weight, the latter affects the effort required for the aircraft in accordance with an exponential law. For these reasons, BMW defined, from the beginning, propulsion as the most practical one which has the relatively lightest specific impulse. From graphs showing the specific-impulse weights for different categories of powerplants as functions of the time of operation, one obtains limits for the time of operation within which a certain category of powerplants is superior to any other, on account of low specific-impulse weights. This is illustrated in Fig. 1 figure (taken from the author's previous publication<sup>1</sup>). This figure as well as the following are based on a flight near to the ground at a speed of 900 km/h.

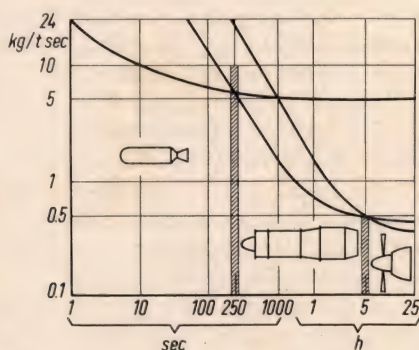


Fig. 1. Specific impulse weights of propulsion plants

In the same manner, among a given category, a certain powerplant type is found to constitute the most practical one for the purpose under consideration, by using curves (Fig. 2) of the specific-impulse weight for different powerplant types within one and the same category. An application of the powerplant thus selected will lead to the lightest and the most economical aircraft or missile.

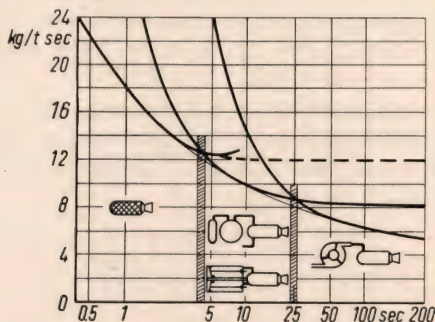


Fig. 2. Specific impulse weights of rocket power plants

It is worth while recalling that more refined methods of comparison shift the time limits to longer periods of operation. This is particularly noticeable in the case of expendable aircraft.

By considering the specific impulse weight for every single manoeuvre during an operational flight separately, and selecting appropriate kinds and types of powerplant to suit each manoeuvre, the most practical combination of two or more different types of propulsion is found; in addition, the relative thrust contributions by component engines can also be established.

### 2.3. Integrated Design and Development Procedure

By tradition, the aircraft designer laid out an initial project for his aircraft before looking around for the most suitable engine. In fact, he generally had to take what the market had to offer.

The same procedure was adopted when the design of guided missiles was taken up. An exception was the development done at the HEERES-VERSUCHS-ANSTALT Peenemünde where a central engineering staff directed a simultaneous development of the airframe, of the propulsive plant and of the equipment to be integrated within the frame of a general project. This manner of approach has since become known as weapons-system engineering.

BMW found very soon that a mutual blending together of airframe and powerplant forms a necessity, in particular when non-conventional propulsive plants have to be incorporated. To achieve and to safeguard such integration, the design of an aircraft or missile must already be perfected in the initial project stage; this needs closest co-operation between the aircraft and the powerplant manufacturer. That such a co-operation was practical within the industry, proved the progressive guided-missile projects which were made by Professor Dipl.-Ing. W. BLUME of the ARADO Works; these projects were, indeed, organic solutions which took into account the properties of the airframe as well as those of the powerplant.

BMW were solely providers of powerplants; but they were eager to effect a similar integration of airframe and powerplant for all other projects. On this reason, a special projecting office was established to furnish complete lay-outs for aircraft and for missiles.

Such complete-aircraft projecting proved in an excellent manner the advantages of complete integration and the most economical use of the effort which was incorporated. This success made it desirable to go a step further, and to extend the activities to actual development work on the integration of airframe, powerplant, and equipment.

As early as 1942, a corresponding development plant for guided missiles was planned at Graz in Styria. This project had, however, to be shelved when it became obvious that results from such development could not be expected to become operational during the war.

### 3. SPECIFIC DEVELOPMENTS

The argument followed above, may appear rather academic.

However, it can be shown from concrete facts and experience that the conception of a quality parameter and of specific-impulse weight (in the same way as equivalent concepts used elsewhere under other terminology) freed the engineer who developed unconventional powerplants, from the rigid and

senseless shackling by "efficiencies" of various character. The new and logical scale of evaluation which had been imposed gave impetus for the planning of special developments in various powerplant categories. Since the powerplant forms the heart of the aircraft or missile, an impetus for developments in the guided-missile field itself followed logically.

A few examples will be given of such planned efforts along new lines.

### 3.1. Supercharging of Piston Engines

The concept of "impulse weight" and of "specific-impulse weight", respectively, refer to the free thrust basically, i. e. to the actual thrust force which is produced by the powerplant for propulsion of the airframe. An investigation of the positive and negative thrust contributions which are supplied by piston engines — disregarding limits imposed by requirements for high thermal efficiency — logically leads to supercharging combined with an inner cooling of the engine cylinder; the inner cooling, though undoubtedly costly in thermal energy, was needed in order to avoid undesirable high peaks in temperature and in pressure.

Within the loading limit allowed by the permissible stresses, the method of adding coolant internally together with supercharging was immediately applicable for engines equipped with blowers for high altitude flight; in such cases, the blower or supercharging compressor was operated at altitudes close to the ground; the internal and expendable coolant agent was simply injected into the intake manifold, and consisted of a water-methanol mixture to keep the water from freezing <sup>16</sup>. Theoretically, the method had been so well prepared that initial experiments on single-cylinder and on complete-engine test stands immediately produced the results which were expected. (Check-tests of this method were made by Professor Dr.-Ing. H. TRIEBNIGG of the INSTITUT FÜR LUFTFAHRZEUGBAU of the TECHNISCHE UNIVERSITÄT in Berlin, and by others.)

This inner-cooling method leads, as known, to an improved continuous-power rating. When correctly adapted (proper relation between the compression ratio and the injected coolant), the result is not only a much more steady running at increased power but also less than the normal wearing-out after prolonged use, in spite of the quite doubled power.

For fighter engines, the method represented by no means a wartime emergency measure due to the lack of more powerful engines but a genuinely progressive feature in the design of high-powered aero-engines.

When fighter aeroplanes operate at full throttle, the speeds of their engines are relatively close to upper permissible limits; in this case, it could be easily shown that the coolant injection was accompanied by a considerable reduction in the consumption, in spite of the poorer thermal efficiency and in spite of the added coolant consumption and also in spite of the much higher specific consumption which was measured on the test bench.

The method which has been described yields — in particular for high-speed flight — the optimum impulse weights which can be obtained from reciprocating piston engines, and hence conveys to aircraft thus equipped, improved performance in speed as well as in range.

### 3.2. Altitude Performance of Piston Engines

The method recalled in the preceding section is restricted to those applications where the supercharging blowers (either mechanically coupled compressors or exhaust-driven gas turbine compressors) permit to increase the compression ratio substantially above the normal values, as happens for instance when flying at low altitudes with the supercharging blowers for high altitudes. The application of the above-mentioned method for an improvement of the altitude performance was impossible, unless the engines were fitted with additional compressors or with higher geared blowers.

Air combat required a boosted altitude performance, i. e. temporary engine-power increases only. In such cases, a variation of the method could be foreseen in which an oxidant provides the supercharging. The choice of such an oxygenating or oxygen-carrying agent determined the nature of the expendable coolant which was needed. Obviously, it was of advantage to inject the coolant together with the oxidant, and the choice of the oxidant as to its concentration, should be in accordance with this.

Consequently, BMW conducted preliminary experiments with peroxyde of hydrogen and with concentrated nitric acid.

Professor Dr.-Ing. O. LUTZ, most probably following the same trend of thoughts, proved more lucky in the choice of an oxidant by selecting nitrous oxide ( $N_2O$ ). This and the corresponding equipment for temporarily boosting the performance at altitude was introduced into service following a surprisingly short development time <sup>16</sup>.

### 3.3. Take-Off Thrust of Turbojets

Compared with propeller-driven powerplants, gas turbine jet engines give no substantial increase of thrust when the speed of flight approaches zero. Hence it was desirable to improve the take-off performance of turbojets, i. e. to increase the thrust during a short period of operation.

It could again be shown that quite sensible thrust increases would result from an injection of coolant, in particular at the compressor entry.

Such internal cooling of the compressor intake produces an increased mass flow and a higher drop at the discharge nozzle, and both factors yield more thrust.

As coolant, a water-methanol mixture was again chosen because of its relatively low freezing point, the exploitation of the methanol as fuel, and because of the higher vapour pressure of the methanol which facilitated the vaporization of the coolant.

For turbojets with air-cooled hollow turbine blades, the injection of waste coolant into the air-cooling ducting promises another possibility for boosting the thrust <sup>17</sup>.

### 3.4. Oxidants in Rocket-Engines

From its inception in 1939, all work done by BMW in the field of rocket-engine development was limited to liquid propellants. The first problem which arose, was the choice of propellants. As the oxygen-carrying agent is of more importance than the fuel, this meant a selection of the most suitable oxidant.

At that time liquid oxygen, at the HEERES-VERSUCHSANSTALT Peenemünde, and highly concentrated peroxide of hydrogen, at the WALTER Works in Kiel were in practical use. Research of that period experimented with liquid oxygen enriched by ozone, and with liquid monopropellants of high energy. At the time, BMW might have chosen one of the two oxidants already in use, and this would have enabled the firm to embark immediately upon the design of new types of rocket-engines.

Being a responsible industrial enterprise, it was, however, not as easy to decide upon the adoption of one of the two oxidants as a basis for a broad development. This was not because one of them was better but because both were afflicted by disadvantages which were deemed unacceptable by the firm.

Highly concentrated peroxide of hydrogen is an unstable agent which would never have satisfied the safety requirements which BMW considered absolutely necessary. Moreover, its sensitiveness to low temperatures required special storage precautions. Its price was relatively far too high. Finally, the use of peroxide of hydrogen would have made necessary the creation and maintenance of large "shadow" factories in order to safeguard the supply in the event of an emergency.

The main advantage of this agent was seen in its excellent suitability as a turbine fuel because of the ideally homogeneous temperature distribution of its disintegration products. This however, applied solely when turbine-driven propellant pumps were needed, i. e. for rocket-engines supposed to run during prolonged operational periods.

With liquid oxygen, the disadvantage is, first of all, its low boiling point.

Compared with other categories of power plants, rocket-engines are by far superior as high-performance propulsive devices for such applications for which operating periods of a few minutes only are required. Such typical short-period applications are, for instance, the interceptor fighter and the anti-aircraft missile.

In both applications, a very short time interval of a few minutes, at most, is available between the warning of an enemy approach and the take-off or launching; during this interval, all preparations such as fuelling, arming, inspecting, etc. must be completed. In view of the growing approach speeds of enemy aircraft, the interval tends to become shorter and shorter. Considering this, it followed logically that the only chance for air defence to become effective, is to possess interceptor fighters and/or anti-aircraft missiles which are permanently ready for action. Technically, this is not possible when liquid oxygen is employed as an oxidant of the propulsion plant.

Disregarding other specific applications of rocket-engines, it was, therefore, hard to see why a weapons system which in itself was quite capable of yielding high quality parameters from the engineering point of view, should be reduced to an overall quality parameter of near zero, by selecting liquid oxygen. On these grounds, BMW rejected as senseless the idea to develop an interceptor fighter or an air-defence missile, or even a launching catapult on the basis of liquid oxygen.

This would, at that time, have meant the adoption of peroxide of hydrogen.

Before arriving at a final decision, a quick survey was made, during a few weeks, of all other chemical possibilities. The reason for doing so originated

from nothing else but consideration of industrial responsibility, and not from an ambition to embark upon basic research so as to open up a radically new development.

Ozone-enriched oxygen had to be discarded for the same reasons as liquid oxygen. Reasons of safety also excluded all high-energy mono-propellants.

It should be stressed here that the statements above do not stand in opposition to those of Prof. WALTER, who spoke of the detonation of peroxide.

Unluckily enough the engineer is not only interested in detonation limits, but also in sensitivity against technical admixtures which could cause self decomposition. Moreover, the peroxide proved to possess a very considerable sensitivity against its own crystals — crystal growth in 86% peroxide concentration begins at 15 °C, which can initiate a spontaneous decomposition.

As sole choice there remained highly concentrated nitric acid of industrial grade. The more BMW occupied itself with this oxidizing agent, the more it became evident that the best oxidant for rocket-engines had been found. Nitric acid is, of course, by no means ideal, and it recommends itself neither for bathing nor for imbibing purposes.

A reasonable degree of engineering, in particular supply and feed systems with freedom from leakages and with automatic sealing, overcame the disadvantages.

It should be recalled that the decision in favour of nitric acid was made just as BMW had concluded all the thermodynamic and chemical calculations, had collected full information on the production and supply of industrial concentrated nitric acid by the I. G. FARBEN concern, and had just performed the initial combustion experiments with this oxidant.

It was at that time assumed by the firm that in view of the great advantages offered by nitric acid, some slight thermodynamic inferiority compared with liquid oxygen — expressed in a difference of the practically obtainable exhaust velocities, or in the impulse of a fuel/oxidant combination per unit of weight, respectively — would be quite acceptable. (BMW did, hence, not simply compare the enthalpy of the propellant combinations concerned but evaluated the technically achievable exhaust velocities, i. e. actual velocities which result after deduction of the heats required for vaporization, decomposition, cracking, and dissociation, and taking into account the technically attainable expansion ratios of the discharge nozzle.) At that time, too, it was anticipated that there would be the necessity of providing for the acid heating or vaporizing tubes within the combustion chamber so as to safeguard the steady combustion as well as sufficiently short stay-times.

Systematic ignition and combustion experiments with various fuels soon proved, however, that using sufficiently energetic initiation of the ignition, nitric acid — fuel combinations burned stable enough by way of self-heating from combustion fumes. This disposed of any need for vaporizing tubes (which had already been prepared). At the same time a refined rating was done of the nitric acid as far as the quality parameter of the complete aircraft or missile was concerned.

During the experiments, it soon became evident that the higher density of propellant combinations which were based on nitric acid not only compensated for the difference in the specific impulse (compared with liquid oxygen) but

also gave superior performance because of its effect upon the basic rocket equation, and because of raised ballistic cross-sectional loading "section density" of missiles. (Higher density increases the mass ratio of the missile; this in turn is a logarithmic function of the velocity at burn-out.)

In this connexion, reference must be made to the 1943 study by HEERES-VERSUCHSANSTALT Peenemünde in which rockets of V-2 type were theoretically compared on the basis of liquid oxygen and nitric acid oxidants, respectively; as a result, it was found that the nitric-acid variant had a much increased range. The comparison yielded an increase in range from 293 km (182 miles) for liquid oxygen (propellant-mixture density of 0.957; specific consumption 5.08 g/kg sec) to 356 km (221 miles) for nitric acid (propellant-mixture density of 1.337; specific consumption 5.36 g/kg sec), i. e. 21.5 % improvement.

In view of such obvious advantages of nitric acid as oxidant, this BMW method formed not only the basis for all subsequent BMW designs but was also adopted for other developments.

It may be recalled that for all subsequent developments in Germany, nitric acid has been exclusively employed.

Apart from the BMW designs for quantity production of the air-to-air X-4 missile of RUHRSTAHL, the SCHMETTERLING anti-aircraft missile by HENSCHEL, the Ju 248 interceptor (JUNKERS development of the Me 163 B which had been equipped with a WALTER rocket-engine), and the home-defence fighter Me 262 HS, other firms and development establishments utilized nitric acid. They were, e. g., the FLAK-ERPROBUNGSSTELLE Peenemünde with their WASSERFALL and TAIFUN missiles, RHEINMETALL-BORSIG with RHEINTOCHTER, and the firm of WALTER with their latest propulsion unit for the SCHMETTERLING of HENSCHEL.

#### 4. GENERAL DEVELOPMENTS

##### 4.1. Aircraft

Again, some examples will demonstrate how fruitful a logical application of the general principles given in section 2 has been found for the development of new kinds of aircraft and missiles.

##### 4.1.1. Load-Carrying Autogyro Rotor

The first development problem with which BMW were confronted in the field of rocket-engines, was a design study for a device to assist the take-off of heavily loaded aeroplanes. So as not to affect the efficiency of the aircraft, the device was to be jettisoned after take-off; it should however be recoverable in order to be re-used, for reasons of economy, and for this purpose a parachute was considered practical.

An analysis of the effort which would be required to recover the device, to repair and to inspect it after use, considering the unpredictable nature of the landing impact, and bearing in mind the cost and the depreciation of the parachute itself, suggested that the method originally proposed was unsatisfactory: the effort referred to constituted too high a percentage of the total effort allotted for the take-off device itself.

Two different kinds of take-off assistance seemed to offer more favourable prospects:

- a) The permanent installation of a rocket engine with pump-feeding of the propellants; this solution was discarded in view of the modifications which would have been necessary to the aircraft.
- b) The substitution of the parachute by an autogyro rotor; the latter was formed by three foldable blades which were attached to the take-off assisting device itself; this was equipped with a shock-absorbing spike for landing.

Calculations and experimentation for a load-carrying autogyro rotor of this kind were done in collaboration with Professor Dipl.-Ing. H. FOCKE; in 1940, the first practical trials followed. These resulted in some flat approaches, apart from many proper pogo-stick landings. Due to work for which higher priority was claimed, the progress with take-off assisting devices had to be deferred; thus work on the load-carrying autogyro rotor was shelved.

#### 4.1.2. *Load-carrying Helicopter Rotor*

The fighting power of military units greatly depends upon heavy arms and equipment which are available in the fighting zone. The transport of this equipment is often impeded by ground obstacles, notably by rivers, lakes, swamps and the like.

Transport across water-covered regions of about 100 m distances is done with the classic methods of engineer corps; water crossings of the order of 50 km are tasks for the navy.

A specific experience gave cause to investigate this problem; this was done by the rocket-planning department then headed by Dipl.-Ing. DREYER. Taking into account the interference from the enemy which must be foreseen during preparations and the crossing itself, disappointingly low quality parameters were evident as well, for engineer-corps methods, as when using the naval equipment then available.

On the positive side, this investigation yielded good scope for the initial project of a helicopter capable of lifting very heavy loads, as for instance, tanks over distances of around 100 km, at speeds of about 100 km per hour.

Since the rotor was to be propelled by rocket engines which were attached to the blade tips, this aircraft would have been very light and simple, considering the state of the technique at that period. Such a rocket-propelled helicopter possessed, for the operational purpose under consideration, the highest possible value of the quality parameter.

#### 4.1.3. *Anti-Tank Missile*

The impending threat from armour could already be foreseen in 1941.

Anti-tank defences relying on sticky bombs exhibited exceptionally poor quality parameters. Becoming aware of this fact, BMW proposed to the German High Command the development of a tele-guided solid-propellant rocket for anti-tank defence.

Exaggerated hopes may have been held then for the "PANZERFAUST" which was already under development. Anyhow, not less than three years after the proposal had been made, the HEERESWAFFENAMT placed corresponding development orders with three firms which were working for the Army; these included the RUHRSTAHL Works. (BMW were not among these firms as being exclusively under the care of the German Air Ministry.)

These developments failed to yield practical solutions in time, and since the requirement had become rather pressing, BMW were called in too; this was at the beginning of 1945.

Just a few weeks later, the firm was in a position to conduct the first firing trials, with the aid of a special trial command in the Dachauer Moor; the trials were entirely successful. The ad-hoc development was a plane-wing aircraft constructed from light-alloy sheet with wire-guidance, spoiler control and direct sighting; it was to be propelled by a twin solid-propellant rocket motor. The initial firing trials had however, been conducted with BMW liquid-propellant rocket-engine of type 109—448 in order to save time, since they served to establish the stability and the control properties of the missile.

#### 4.2. Power Plants

A few examples may show that the basic design of power plants can benefit well from the guiding development principles which apply to complete aircraft and missiles.

##### 4.2.1. *Marine Torpedo Powerplant*

Successive improvements of naval defence caused the quality parameter for submarines and for torpedo-carrying aircraft to sink persistently whilst the war went on.

To restore these parameters to reasonable values, an improvement of the marine torpedo was indicated so as to give higher underwater speeds and greater range.

Hence the problem arose of finding if, and to what extent, a new kind of powerplant could reduce the impulse weight against that of the classic torpedo engine.

Under given conditions, a gas-turbine driving a water-propeller should prove most economical when it would work with a gas/steam mixture generated from a near-stoichiometric combustion of hydrocarbons using concentrated nitric acid under addition of coolant, the latter consisting of sea-water. Proposals to utilize sea-water in a similar way for submarine propulsion had been made before; but such solutions had not been successful in view of the salt deposits on turbine blading, with subsequent deterioration in efficiency, etc. on account of blade-section deformation. BMW felt, however, that using a high pre-heated coolant (because of the very short period of operation — of the order of minutes) would render any salt deposits and their effects practically of no consequence.

Basic laboratory investigations seemed to confirm this view. Consequently, combustion experiments were conducted on rocket-engine test-stands with gas/steam generators using sea water. The results led to the design of a corresponding torpedo powerplant of 500 b. h. p.

#### 4.2.2. Rocket Engine with Programmed Thrust

For the development of an air-to-air guided missile (type X-4 of RUHRSTAHL, designed by Dr.-Ing. M. KRAMER), a relatively high short-time thrust was demanded for initial acceleration, in addition to a lower cruising thrust for the major part of the trajectory. Considering the operating time and the purpose, a twin solid-propellant rocket would have seemed practical (see section 2.2). However, at that time neither twin rocket-motors were available nor any solid-propellant rockets of 20 sec burning time; no promises could be obtained from the development authorities concerned if and when such propulsion units would become available.

It fell, hence, to BMW to develop a liquid propellant rocket-engine for this missile, satisfying not only the requirements stated above, but also possessing high reliability, plain construction and suitability for simple manufacture. The result was BMW 109—448 shown in Fig. 3.

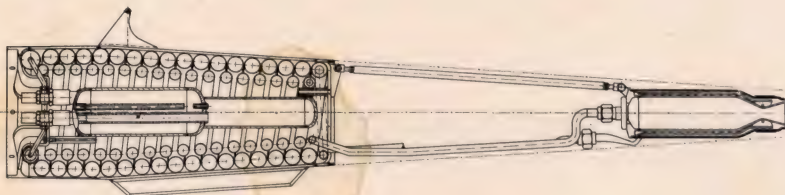


Fig. 3. Engine BMW 109—448

Characteristic for this type is the absence of regulating devices and of other armatures in the propellant supply. The successive variation of thrust was effected through the design of a pressure-gas container of appropriate size and initial pressure, by proper selection of injection nozzles in the combustion chamber, and by accurate dimensioning of the nozzle-throat area.

Further details concerning this engine type which may be of interest, will be discussed in Sect. 5.

The following development data of this engine may be quoted here:

First conference for the development was in early summer 1943; the work order was issued in late summer 1943. About 1500 engines were built by February 1945, 1300 of them in the plants in Stargard, which were bombed then and about 250 engines by the BMW development plants.

So it took only 17 months from the work order issue for the development to the production in a new plant (the GERÄTEWERK in Stargard), including not only the projecting and developing, the testing on conventional stands and special test stands, the flight tests and the investigations of production particulars, but also the building and installing of a new plant. This was indeed a good performance of the entrusted personnel.

#### 4.2.3 Pressure-Fed Rocket-Engine without Pressure Gas

For a guided-missile design (the Hs 293 glide-bomb of Prof. Dr.-Ing. H. WAGNER) an auxiliary rocket-engine propulsion having about 12 sec burning-time, was designed in order to extend the range of the missile. This engine had

propellant feed by compressed air and ran with peroxide of hydrogen. Operationally, the propulsion of this weapon proved rather deficient on account of internal icing up. This happened in spite of using special pressure-regulators and dry compressed air, besides, weapons suffered deterioration during storage after they had been prepared for action, on account of decomposition of the peroxide of hydrogen. Consequently, the quality parameter of the complete aggregate (consisting of the carrying aircraft and of the weapon) had become rather poor.

BMW was asked to develop a powerplant free from such deficiencies.

The internal icing was linked to feeding by gas pressure, whilst the other trouble could be cured by adopting nitric acid as oxidant. BMW did not feel that the gas-compressor plants with which the front-line military formations were equipped, were reliable enough for the supply of really dry air; and the development of pressure regulators which were able to cope with compressed air of industrial moisture content had not been successful as far as the specialist firms were concerned. New development was not considered warranted.

Consequently, BMW accorded preference to propellant feed without any air-pressure, and succeeded in providing constant thrust with the aid of an engine which was devoid of gas-pressure regulators, and of armatures in the supply lines. Equipped with a differential piston device, the BMW rocket engine type 109—511 resulted; this shown, as an outline in Fig. 4. Interesting details of it will be further discussed in section 5.

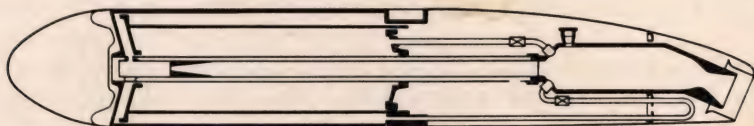


Fig. 4. Engine BMW 109—511

The principle of differential pistons guaranteed a steady thrust force which could be varied within wide limits solely by using injection nozzles of different gauges. Thus, for instance, one and the same prototype engine (constructed for basic trials within a period of a few weeks) was able to run with functional combustion-chamber pressures between 10 and 250 atmospheres absolute.

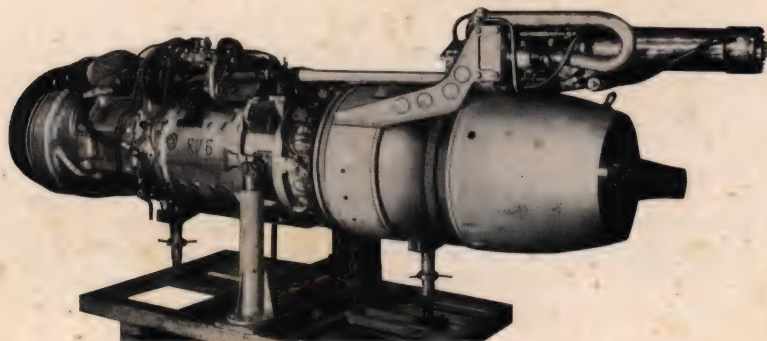
#### 4.2.4. Combination Powerplant

BMW established from performance calculations for the Me 262 that, for home-defence, a combination powerplant composed of a turbojet and of a rocket-engine, would greatly enhance the tactical value of the aircraft. The authorities decided to develop a home-defence variant Me 262 HS on this basis. WALTER and BMW were charged to design a corresponding special powerplant. For the rocket-engine pump feed of the propellants could alone be used, in view of the prolonged period of operation (see section 2.2). The WALTER firm provided the rocket-engine which had been developed for the Me 163 B; this was installed in the tail end of the Me 262 fuselage. BMW was more progressive and wanted to introduce a simplified and lighter equipment which should, at the same time, be more reliable. The turbine-pumps and their

fuel, steam generators, regulators etc. were therefore abolished; instead, the propellant pumps were driven by the turbojet.

In collaboration with Dr.-Ing. H. OESTRICH who developed the BMW 003 turbojet in his development establishment, the variant BMW 003 R was created; this incorporated the BMW 109—718 rocket-engine.

In this manner, a new kind of combination powerplant had come to life, with a number of typical advantages. (The first combination powerplant appears to have been the piston-engine equipped with exhaust gas-turbine supercharger.) Fig. 5 shows this powerplant. Details will be discussed in section 5.



*Fig. 5. Engine BMW 003 R with BMW 109—718*

It may be mentioned here that the propellant pumps were mechanically driven, and that the rocket-engine was started and stopped by electric actuation of a lamellated coupling. A mechanical coupling had to be preferred, for the reason that it could be made available in the shortest possible time. Hydraulic coupling has the advantage that it does not transmit vibrations; drives using compressor air for pump turbines, transmit no vibrations and also give more freedom in the installation of the rocket-engine itself. However, these two alternatives would have needed more time for development.

## 5. ROCKET-ENGINE DEVELOPMENT

That the general development principles apply, too, to details of development, will be shown by discussing some examples taken from the BMW rocket-engine development. It may be emphasized one again that an aircraft must be designed around its powerplant, and that lacking a suitable powerplant, it becomes useless. Hence the highest quality parameter of an aircraft or missile is limited by the impulse weight of its powerplant. The actual value of the quality parameter, however, depends upon the attainable reliability of the powerplant. The reliability of the airframe and of the equipment, of course, influence the quality parameter too.

It has to be stressed that the following chapters are not restricted to the use of nitric acid alone, but that they touch fundamentals of the rocket engine construction.

## 5.1. Methods

These considerations lead to a selection between applicable functional methods to ensure maximum reliability as well as lowest weights and lowest (direct and indirect) operating costs.

### 5.1.1. *Fuels*

When BMW had established (see section 3.4) that nitric acid formed the most suitable oxidant, it had to be decided if there was something which could be deemed the most suitable fuel. This problem proved surprisingly much more complex than the selection of the oxidant. It was some time before BMW arrived at a definite solution.

In the beginning, methanol seemed preferable because a mixture of methanol and nitric acid was easy to ignite, burned well and was easily obtainable in large quantities. BMW obtained during endurance tests with methanol/nitric-acid mixtures, at 35 at. absolute combustion-chamber pressure and at sea-level, specific consumptions of 4,80 g/kg sec, corresponding to an exhaust velocity of 2040 m/sec, i. e. a specific impulse of 208 kg sec/kg.

On the other hand, theoretical studies indicated the relative importance of the density of propellant mixtures upon the quality parameter; this gave cause to accord preference to liquid hydrocarbon fuels (see also section 3.4). Finally, the desirability was foreseen — especially for the case of auxiliary or additional rocket propulsion, such as that for Me 262 HS — to reduce the supply to oxidant alone, by burning ordinary aero-engine fuels.

The choice of hydrocarbons thus became a clear aim for the subsequent research. The latest development of BMW paved the way for such change (see section 5.14; in particular, also sections 5.1.2 and 5.1.3).

### 5.1.2. *Igniting Agents*

As mentioned, methanol was the originally adopted fuel; this gave safe and reliable ignition when glow-plugs and pyrotechnic igniters were employed, and even with low-energy spark-plugs, ignition could be obtained.

A simplification in the design of rocket engines with a simultaneous increase of safety was possible by introducing self-igniting agents; initially well-known organic metal compounds such as zinc ethyl were adopted. Efforts were made to render these agents sufficiently inactive against atmospheric oxygen, without impairing their igniting qualities when coming into contact with concentrated nitric acid. Although it appeared during the early stages that this method was realizable, all inactivated mixtures suffered so much from aging when in contact with the free atmosphere that they were soon incapable of reacting with nitric acid. This development did not refer to ignition in rocket-engines alone, but, from the beginning, to high-energy ignition for turbojets too, in particular for the purpose of re-kindling the combustion in flight.

Taking a hint from references in text-books of inorganic chemistry, the ignition behaviour of nitric acid in contact with turpentine was investigated. The first experiments were disappointing since fuming and smouldering always occurred, but seldom ignition.

Spectral analysis of the tests during which ignition occurred, consistently indicated that traces of copper were present. Adding powdered copper to the turpentine, the first reliably acting ignition agent was obtained which remained unaffected by atmospheric oxygen.

This discovery gave a mighty impetus to progress during further experimentation. After a few weeks work, the most prospective agents among the amines had been laboratory-tested, by letting droplets fall into cups with nitric acid; the most reactive substances among them were then more closely investigated in testequipment for ignition-delay at normal temperatures and at low temperatures.

Ignition agents which reacted with concentrated nitric acid were collectively termed "TONKA" by BMW; in general, they were amines. During the transition period "TONKA" agents were also employed as fuels. "TONKA 250" was a fifty/fifty per cent by volume mixture of raw xylidine and triethyl amine and had an ignition delay of  $2.1$  to  $2.5 \times 10^{-2}$  sec during drop tests at room temperature, and a delay of about  $1.5 \times 10^{-2}$  sec in contact with a copper-salt catalyzed concentrated nitric-acid. It proved thus more eager to inflammation than BMW had ever hoped for, and it represented, therefore, an excellent ignition agent for nitric acid.

Motivated by similar ideas, Prof. Dipl.-Ing. O. LUTZ and his collaborators had searched, before BMW began their investigations, for self-igniting propellant systems. He had discovered for instance the well-known system of peroxide-of-hydrogen/hydrazine-hydrate; such systems were termed by him "hypergole". The same institute discovered, subsequently, ignition agents for nitric acid and termed them "visole" (essentially a vinyl-ether basis); the behaviour of the visoles was about the same as that of the BMW "TONKAS".

Subsequently, BMW tended more and more to restrict the use of "TONKA 250" as ignition agent and as self-reacting propellant mixtures solely to the period during which the development needed speeding up (see section 5.1.1). One important reason for this attitude was the increased fire danger under enemy action and in crashes, especially for piloted interceptor fighters when hypergole propellants were used.

### 5.1.3. Safety Installation System

The first BMW rocket-engine designs were the type 109—511 for an air-launched guided missile, and an aircraft-propulsion unit type BMW 109—510, which was to be installed into the JUNKERS 248 modification of Me 163 C. It was necessary to design these engines for easy and safe handling in operation as it would have not been possible in service to safeguard the functioning of the engine by relying upon a manual sequence of appropriately timed control actions. All BMW rocket-engines which were running on normal, i. e. not self-reacting propellants, incorporated the following typical safety arrangement:

Actuation of the starting lever by the pilot first put the ignition into action; once the ignition was working well and safely, an automatic device caused propellants to flow into the combustion chamber to start the combustion.

On the BMW 109—510 rocket-engine, a photocell registered the presence of an ignition flame before the electrically actuated propellant valves opened. Some trouble arose from soot which formed on the photocell so that the latter

failed to register even when the engine ignition functioned properly. Besides, the effort in electric equipment had become rather too elaborate. BMW therefore, substituted for the photocell, a pressure-sensitive relay which reacted upon the pressure increase which accompanied the functioning of the ignition (see also section 4.1.4).

The rocket-engine type BMW 109—511 had a different safety-relay arrangement whereby the smoke fumes produced by the two pyrotechnic igniters during about 5 sec burning time directly supplied the pressure which was needed for the introduction of initial quantities of the propellants into the combustion chamber. This again gave complete safety.

#### 5.1.4. *Pre-Combustion Chamber for Ignition*

The starting of rocket-engines needed a relatively energetic and powerful ignition flame in order to initiate, in a reliable and rapid manner, the inflammation of a propellant mixture which was still poorly adapted to combustion when the components began to flow into the chamber. In order to exploit the pressure of the ignition flame for monitoring the propellant supply (see section 5.1.3), it was found practical to produce this igniting flame within a special pre-combustion chamber. Thus, for a given throttling, a relatively low quantity of ignition agent gave sufficiently high ignition pressures by which the hydraulic operation of the propellant valves could be effected. This meant that the electric relays needed for the actuation of the valve could be discarded. Besides, a pre-combustion chamber gave a possibility of reducing the ignition-delay below the delay times which had been found during the droplet tests (see section 5.2.2).

#### 5.1.5. *Thrust Regulation*

Considering the impulse weight, rocket-engines are best when running at full thrust; this makes arrangements permitting convenient starting and stopping attractive. Reasonably good impulse weights can also be obtained when two or three combustion chambers are arranged in such a way that they can be started or stopped singly or combined. When employing such step-wise thrust regulation, the provision of three different-sized combustion chambers would give eight variations of the thrust.

For the interceptor engine 109—510 as well as for the missile-propulsion 109—558, such stringent requirements for thrust regulation were imposed that a relatively simple step-wise regulation would have been unable to satisfy them. Since the 109—510 engine had propellant feed from turbopumps, a by-pass regulation of propellant flow presented itself as a practical solution; this meant that the turbines and the pumps could be run at constant speeds.

The BMW rocket-engine 109—558 for the SCHMETTERLING guided missile, shown in Fig. 6, had however been designed for gas-pressure feed of the propellants, in view of the short period of operation; this forbade a by-pass regulation which would have required the return of excess propellant into the pressurized tanks.

BMW therefore, effected the thrust regulation by providing two flat sliding valves within the nozzle heads. These could be adjusted, through gear wheels, from an electromotor which had a reversible sense of rotation. The motor again

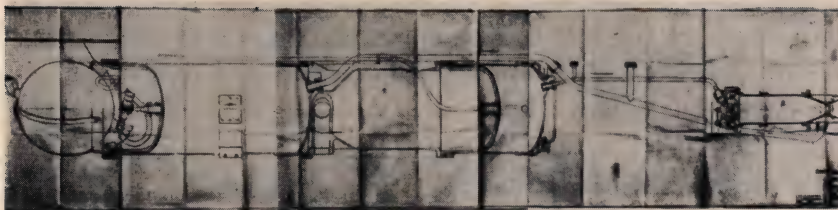


Fig. 6. Engine BMW 109—558

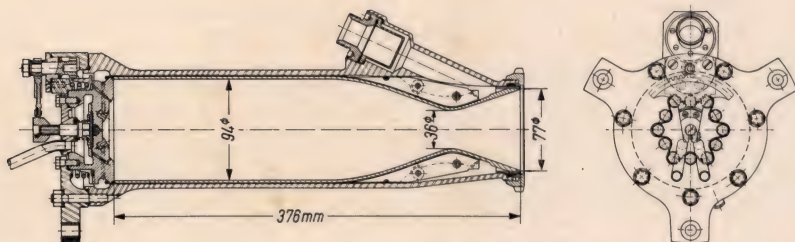


Fig. 7. Combustion chamber of engine BMW 109—558 with thrust regulation head

was monitored by a MACHmeter so that, following an initial accelerating period, the missile flew at a given constant MACH number independent of its attitude. Fig. 7 shows the method of thrust regulation.

#### 5.1.6. Delay Times

The relatively wide range within which the thrust of this propulsion plant could be regulated (in the case of BMW 109—558, a ratio of 1:10) caused a disagreeable surprise. The thrust regulation itself proved, indeed, successful from the outset, and even the propellant flow regulating slide exhibited not the slightest tendency to seize up as had been feared since it was merely "lubricated" by nitric acid. (The necessity to grease with paraffin wax or graphite had been anticipated as an eventually-needed remedy.)

The critical problem of combustion-chamber cooling when running at low thrust coefficients had been solved, too. But it was experienced during tests that the total impulse of the engine dropped at small thrusts below the calculated and guaranteed values; this seemed, at first, quite inexplicable.

BMW had assumed till then that if the reaction was to be complete it was necessary for the propellant fumes to remain in the combustion chamber for a certain time, but that this delay-time was, for a given combustion-chamber shape and size, independent of the thrust; the combustion chamber of this engine had been dimensioned accordingly. A more refined investigation proved, however, that the delay-time, required for efficient combustion, is a function of the combustion-chamber pressure and, hence, longer when this pressure decreases. It thus became evident that the original combustion chambers for this engine had been insufficiently dimensioned, in view of the necessity to provide for low thrust coefficients.

## 5.2 Design Properties

### 5.2.1. Propellant Nozzles

Discussion here will be restricted solely to injection nozzles for normal propellant systems (see section 5.1.1), i. e. for propellant combinations which are not hypergole. (Propellant nozzles for hypergole systems are compromises between a propellant nozzle proper and an ignition nozzle (see section 5.2.2) or more precisely, a combination arrangement of an igniting nozzle with propellant nozzle.)

Requirements for propellant injection nozzles are:

- a) good preparation for mixture and combustion of the single propellant sprays at lowest possible injection pressure;
- b) good intermingling of the sprays of the propellant components;
- c) Low sensitivity under action of fluctuating pressures within the combustion chamber;
- d) decreased input at less than full build-up of the combustion-chamber pressure, i. e. when the pressure differential between injection and combustion chamber is greatest.

All these requirements have to be satisfied when the consumption is to be low and close to the theoretically predicted values if the operation is to be free from vibration, and if the minimum design weight is to be attained. Simple swirl nozzles gave very good distributions in the spray. These nozzles were designed in closed form so as to reduce initial pressure rises (e. g. the 109—511 rocket engine). Good intermingling between the component sprays was achieved by using several nozzles for each component of the propellants. So as to get optimum consumption with small volumes of the combustion chamber and to ensure easy cooling, double swirl nozzles were developed. (Insufficient mixing causes, even when running with excess fuel, hot oxygen to reach the walls of the combustion chamber; this would demand excessive cooling in order to prevent deleterious reactions.)

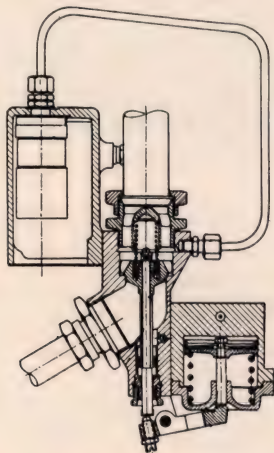
A low sensitivity under action of pressure fluctuations within the combustion chamber, i. e. a negative acceleration of the propellant jets which are introduced, was effected in two ways: The nozzle was designed as a closed unit and swirl throttling devices were provided in the propellant supply lines; these devices could also be organically incorporated in the nozzles themselves.

To avoid an occurrence of pressure peaks when the combustion was initiated, the most reliable method seemed to be a steady variation of the cross-sectional area at the nozzle supply side. This typical BMW design feature proved its efficiency from the beginning when the first take-off assisting device was under development; the feature was retained for subsequent developments of propellant injection methods.

This method of obtaining a steady propellant-flow regulation was also applied for the design of different kinds of nozzles. Fig. 8 illustrates the valve unit for perforated-nozzle heads of the 109—718 rocket engine.

The release of the fuel under pump pressure in the engine ready for operation is done by the piston, which is visible on the right side of the figure. The piston receives the fuel on the upper side by a needle valve (not shown) via a blind

working as throttle. The adjustment of the throttle to the section of the piston enables the regulating of the time of the valve operation.



*Fig. 8. Valve block of engines BMW 109—510 and 109—718*

For the 109—558 rocket-engine, the method was varied by keeping the regulating slides in their minimum-thrust position during the starting; thus, a single nozzle group only came into play instead of the twelve groups which were injecting when the engine was running at full thrust.

Adopting closed, steady regulating, coaxial twin-swirl nozzles for the propellants, this development was considered as concluded. BMW subsequently intended to standardize such twin nozzles in two thrust sizes, for 250 kg thrust and 1,000 kg thrust, respectively.

### *5.2.2. Nozzles for Ignition Agents*

The main requirement for ignition-agent nozzles is to produce a reliable and sufficiently energetic source of ignition. Another requirement is that this source of ignition comes into action with the least possible delay. It was interesting to discover that the ignition delay is the greater, the finer the agent is distributed. BMW, for instance, obtained smoke without any formation of flame in spite of excellently hypergole systems as soon as intentionally high injection-pressures were applied.

This phenomenon is less surprising if the mechanism of ignition and of combustion is subjected to closer investigation. Very finely distributed droplet mists of hypergole systems in which single droplets come into contact, react with each other by mutual diffusion over the contact area. The hot fumes which are subsequently produced from the reaction, cause pressures which are greater than that of the surroundings; this causes the two droplets to separate again before the reaction sufficiently progresses to flame formation and this terminates the reaction whilst it is still in the initial stage. The droplet separation follows the quicker the smaller and lighter the propellant droplets are.

Hence it became obvious that some compulsion is needed to force and to keep the droplets fused together and to enable them to overcome the pressure exerted by the fumes. This provided the principle underlying the design of efficient ignition-agent nozzles.

The compulsion needed could, firstly, be obtained from exploiting the impulse of the injected propellant jets. In this way a very efficient ignition nozzle was formed by a three-hole arrangement with the centre bore for the fuel and the two outer bores for the oxidant; the three holes were in the same plane but so that the outer jets were inclined towards the axis of the central one; the injected sprays were, therefore, impinging upon each other.

The 109—448 rocket-engine had to satisfy especially severe requirements with respect to starting. In this case, BMW exploited the walls of the combustion chamber for further reduction of the ignition delay and thus a reduction of combustion-chamber pressure peaks.

The hypergole fuel was injected a few tenths of a second in advance of the nitric-acid oxidant; the combustion-chamber walls were thus coated with a film of ignition-producing agent; when the nitric acid impacted upon this film on the walls, large areas of ignition were produced. All this was very simply effected by a simultaneous release of the two propellants. Since the acid had first to fill the initially empty cooling passages, the desired delay in the oxidant injection was a consequence.

Although the three-hole nozzle proved entirely satisfactory in operation, it put too much strain upon the manufacture. The bores had to be accurately in one and the same plane and to be cleanly rounded. Moreover, the nozzle produced increasing ignition-delay when the injection pressure was raised because the propellant jets began to break down into mist before meeting. BMW proceeded, therefore, to improve upon the ignition nozzle design by deriving the necessary impact energy for the droplets from centrifugal forces. To effect this, the constituents were injected tangentially inside a tubular section. This design, indeed, gave the minimum ignition-delay at all injection pressures, and it was easy to make.

### *5.2.3. Heat-Absorbing Combustion Chamber*

When investigating the wall heating resulting from unsteady heat-flows<sup>18</sup>, it was seen that up to about 15 sec of burning a methanol/nitric-acid mixture, a combustion chamber of light alloy was practical without any other cooling than the heat absorption capacity of the walls.

The BMW 109—511 rocket-engine which has already been shown, had such a combustion chamber cast from silumin. This proved a very light component and was also very easy to produce.

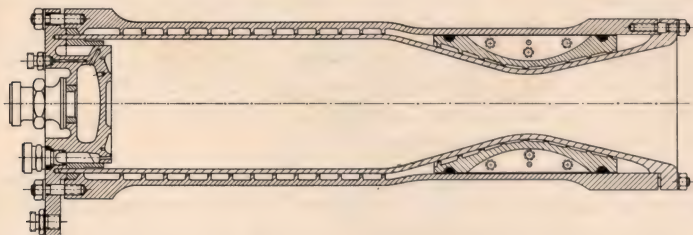
### *5.2.4. Liquid-Cooled Combustion Chamber*

For longer burning times, liquid-cooled combustion chambers are necessary. All BMW types of this kind used the nitric acid as coolant.

A reliable cooling by way of enforced convection requires relatively close tolerances in the cooling passages. When BMW began their development, all combustion-chambers known were welded fabrications. Such components were

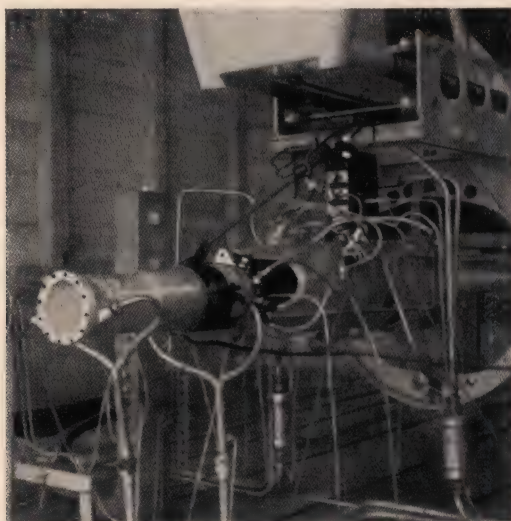
not only expensive in production but constituted nearly artistic masterpieces when duralumin was used. (For such a combustion chamber, duralumin offered sufficient immunity against action of the nitric acid.) Besides, the final inspection proved neither straightforward nor reliable enough.

BMW arrived, therefore, at a combustion-chamber design which has since become typical for BMW rocket-engines. This was built up from single, machined components which were simple to make, simple to inspect, and easy to assemble or to dismantle.



*Fig. 9. Combustion chamber*

A divided filling piece was provided around the throat of the combustion chamber to define the coolant duct. Fig. 9 shows one of the latest types of this combustion-chamber design, in this case for the 109—718 rocket-engine. This figure shows the typical construction formed by an outer and inner combustion-chamber wall with the filling piece and nozzle head. The last type had the coolant duct formed as a slot.



*Fig. 10. Arrangement of combustion chamber test stand*

The external form of this combustion-chamber construction can be seen from Fig. 10 which shows the typical BMW experimental arrangement of combustion-chambers and powerplants.

#### *5.2.5. Gas/Steam Generator*

For rocket engines with propellant feed by turbo-pumps, peroxide of hydrogen might be used (see section 3.4). However in order to avoid any deterioration of the quality parameter which would arise from impaired immediate operational availability, the turbine should function with the propellants themselves.

The final aim was to utilize directly gas bled from the combustion chamber; this would have to be sufficiently cooled, by injection, before reaching the turbine blading. (Combustion temperatures of  $3,000^{\circ}\text{K}$  are common, whilst the turbine blades would stand no more than  $1,000^{\circ}\text{K}$ .) However, in order not to hold up the general progress, the initial development adopted a separate gas/steam generator distinct from the combustion chamber, but resembling it in design. Inside the generator, a methanol/nitric-acid mixture was burnt in about stoichiometric ratio. Behind the flame zone, the hot gases were cooled through the injection of coolant by swirl nozzles; as coolant agent a water/methanol mixture was originally employed. When the aggregate of gas/steam generator, turbine, and high pressure pump (for the 109—510 A rocket-engine) functioned properly, the wish to perfect a genuine two-propellant operation prevailed; moreover, new combustion chambers were designed to burn hydrocarbons, and no longer for the use of methanol fuel. A separate coolant could be substituted either by excess nitric acid or by excess fuel. Nitric acid proved very well suited for such a purpose. At fume temperatures exceeding  $620^{\circ}\text{C}$  not even nitrogen dioxide was found present in the exhaust from the turbine.

As in the case of peroxide of hydrogen, the presence of free molecular oxygen did not prove a disadvantage in any way.

#### *5.2.6. Propellant/Pressure-Gas Separating Pistons*

Any residue of propellant which remains in the propellant tanks at burn-out, means an increase of the impulse weight. For aircraft propulsion plants, e. g. the types BMW 109—510 and 109—718, a pendulum-type discharge reduced the unusable quantities of propellants to an acceptable minimum.

In the case of guided missiles, need for complete insensitivity to accelerations forms an additional problem. This insensitivity combined with the complete use of the propellants stored was accomplished by BMW through the provision of pistons within the compressed-gas feeds, so as to separate the propellant liquids from the pressure gas. (Special problems to seal the pistons do not arise in this case since the liquid and the gas are at practically the same pressure.)

Separating pistons were provided too for the types 109—448 and 109—558 (Fig. 6).

Since the 109—448 rocket engine had, for reasons of production, propellant tanks which were formed by spirally arranged tubes, the separating pistons had to suit the form of the container. The pistons were rather long, possessing at their ends spherical guiding pieces; the space taken between these end-pieces was fitted with several plastic discs, with distance pieces between them, to give flexibility.

This piston design offered aeromechanically the additional advantage of giving perfectly steady changes in the position of the centre of gravity, and moreover, a centre-of-gravity travel which could be kept as small as desired.

The 109—511 rocket engine also possessed, by virtue of its separating pistons, insensitivity to accelerations, and centre-of-gravity positions which were well defined at any instant.

An attempt was also made to separate the liquid propellants from the pressurizing gas by way of elastic diaphragms. During 1944, plastic cells coated with a homogeneous aluminium film became available, for giving perfect sealing and immunity to the action of nitric acid. This permits the simplification of the design of propellant-feed systems.

#### 5.2.7. *Starting Pumps for Propellants*

Safety precautions demanded the provision of an adequate initial starting pressure when the high-pressure feed-pumps were used. The starting pressure had to be adopted to the vapour pressures of the propellants, and to the acceleration of the aircraft. Fuel-tank pumps then in use for aircraft were far too heavy, too big, and also too expensive, considering the deliveries required per second by rocket-engines.

For this reason, BMW developed starting pumps of their own, which utilized the jets supplied by the high pressure pumps.

#### 5.2.8. *High-Pressure Pumps for Propellants*

Available high-pressure fuel pumps, too, could not be adopted when BMW took up their rocket-engine development because of their weight, bulk and expensive construction.

In collaboration with HENSCHEL, BMW, therefore, developed simple, single-stage centrifugal high-pressure pumps, which operated at the speed of the driving turbines, i. e. at 16,000 to 20,000 r. p. m. Because of their simplicity and reliability, similar pumps were also adapted for the feeding of ignition agents in rocket-engines and in turbojets, and also as coolant pumps for supercharged spark-ignition engines (see section 3.1).

To give an idea of this development, it may be said that a pump aggregate for a rocket-engine of 1500 kg thrust to feed at 50 at absolute injection-pressure, weighed less than 5 kg, and had a maximum pump diameter of about 120 mm.

## 6. SUMMARY

In this paper an attempt has been made to describe the principles behind the development of the various aircraft and powerplants produced by BMW. In connection with this, a few examples were discussed to demonstrate that even today these principles possess more than limited value, and that component developments, as for instance those for functional groups in the construction of rocket-engines, are dominated by the very same principles.

Since there might still be a disinclination to accept fully the clear implications which logically follow from the adoption of the quality parameter, which was defined in section 2.1, the author here states some of the most obvious conclusions:

- a) A weapon having a quality parameter of the order 1, is absurd since its war-like application against an enemy of about equal strength amounts to masochistic self-mutilation, and against a stronger enemy, to suicide.  
The development of equipment which has a quality parameter of 1 or less should obviously be neither encouraged nor subsidized.
- b) The evaluation of a diversity in solutions for one and the same problem in weapon engineering should be done solely on the basis of the quality parameter.
- c) Existing equipment should only be replaced if and when the equipment which is to be substituted for it has a substantially higher quality parameter, bearing in mind the effort which is required to institute a new production.
- d) A new development of weapon systems should be carried through only if the result promises a quality parameter which is so much higher that the new weapon system would remain more efficacious even when it is taken into account that the existing weapon might be still further improved. If this is so, the new development deserves to be carried through without any delay.
- e) The use of a universally applicable characteristic parameter provides opportunity for planning authorities to compile statistics upon which long-term planning can be correctly based. This guarantees steadiness in the development policy and help in the economic relations with the industry concerned.

In conclusion, it must be stated that this report on developments carried out by the Rocket-Engine Department of BMW constitutes by no means anything like a comprehensive survey of the entire development effort of the firm. BMW did not occupy themselves only with special propulsions but also developed piston-engines and gas-turbine jet-engines. The references are, therefore, quite insufficient as far as information on the entire work of the firm is concerned. In addition, it is fair to add that more recent developments in the field of rocket-engines as well as in the field of missiles and aeroplanes must be deemed logical consequences of the lines of development indicated by the effort of BMW.

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## DISCUSSION

Prof. Dr. QUICK (Aachen): In your paper you made a distinction between fuel nozzles and ignition nozzles. Could you give any further details, especially on the rules which consequently govern the construction of a rocket combustion chamber?

Dipl.-Ing. VON ZBOROWSKI: The requirements which fuel nozzles and ignition nozzles have to meet are so different that they lead to diametrically opposed construction principles. I only need to recall for instance that good fuel nozzles have to provide for a thorough disintegration of the fuel, whereas good ignition nozzles should not.

A good combustion chamber should not only give low fuel consumptions, it must be reliable and as light as possible, in other words it must give a smooth start without considerable pressure maxima. The pressure maxima when starting were to be kept much lower than the operating pressure in the combustion chamber. In connection herewith the bursting-charges occurring when starting old type combustion chambers may be recalled. These noises were proof enough that the combustion chamber constructions were inadequate.

A good combustion chamber should be fitted with fuel nozzles as well as with an ignition nozzle and the appropriate safety circuit, and this absolutely independently of the use of normal or hypergolic fuels. Contrary to former opinions it may be stressed here, that the use of hypergolic fuels, as is known at present, neither simplifies the construction of a modern combustion chamber nor does it permit the omission of the safety circuits.

Admiral FAHRNEY (Philadelphia): In your lecture, Mr. VON ZBOROWSKI, you stated that a test using nitric acid was carried out successfully in the V-2 and that a greater range was achieved than with the alcohol-oxygen combination. If that is true, why did not the V-2 use nitric acid?

Dipl.-Ing. VON ZBOROWSKI: I would have preferred to have this question answered by the gentlemen from Peenemünde, for instance by General DORNBERGER. All I can say is that it was argued that the switch over to a better V-2, which means a V-2 operating with nitric acid instead of liquid oxygen, would have caused delays in its going into action. Such an argument had to be accepted by higher headquarters, because actually all the Peenemünde experiments were based on the use of liquid oxygen exclusively.

This argument cannot stand for further developments, because these can be based on the better principle right away.

Anyhow, there were no technical reasons against the use of nitric acid. The frequently expressed opinion, that nitric acid is "difficult" to handle, is ridiculous, considering that chemical industrial plants like the I.G. or the DEUTSCHE REICHSBAHN had handled nitric acid as industrial bulk product for years without accidents. There was also experience available, which BMW, the anti-aircraft artillery experimental headquarters and also WALTER (Kiel) had made with rocket power plants.

The Peenemünde comparison experiments, which I quoted in my paper, were based on fuel compositions of gasoline and Diesel oil. May I recall here that these experiments considered a specific fuel consumption of  $5.36 \times 10^{-3}$  kg/kg sec for the nitric acid propulsion of the V-2. If the actual amounts obtained by BMW in their continuous experiments had been used for comparison, their low consumption figures would have proven a much higher superiority of the nitric acid system of the V-2.

Mr. A. G. THATCHER (Denville, N. J.): After hearing Prof. WALTER on hydrogen peroxide rocket developments and Dipl.-Ing. VON ZBOROWSKI on acid rocket developments, it appears that rockets using both types of oxidizers were installed and flown in piloted aircraft, such as the Me 163. I would like to ask if sufficient experimental or tactical flight experience was obtained with these rocket powered craft to draw any significant conclusions as to which oxidizer proved to be the safest and most practical for this type of application. It would also be helpful to hear more discussion on the hazards and practical operating problems encountered in these manned rocket powered flights.

Dipl.-Ing. VON ZBOROWSKI: Although until 1945 most of the practical experience with piloted aircraft had been obtained with peroxide (WALTER-engines), considerably less experience with nitric acid (BMW-engines)\* and none with liquid oxygen\*\*, it can be stated as granted that the engineers with the firms WALTER and BMW were at that time able to furnish power plants with the necessary degree of safety (in accordance with the then relatively small requirements of the office for these developments). The BMW-engines, for instance, have been subjected to many hundreds of combustion periods in the same combustion chamber and to uninterrupted operation periods of several quarters of an hour while passing the type tests.

\* This development started several years later.

\*\* In spite of such projects being worked on at the Heeres-Versuchsanstalt Peenemünde.

The practical differences between the various types of engines result solely from the chemical and physical data of the oxidizers: liquid oxygen power plants do not come into consideration due to the start preparations. To try to keep such an engine ready for start for a while by refuelling liquid oxygen, would result in icing the whole power plant in a short time. (The air humidity turns into ice on the engine. Since air humidity remains practically constant, the ice layer increases continuously.) Peroxide engines and nitric acid engines show almost the same performance results, the latter being slightly superior, since nitric acid gives a somewhat better consumption figure. The boiling points of both — peroxide and nitric acid — are high enough for summer- or tropical operations; for winter- or arctic use both peroxide and nitric acid are superior as well for operation as for supply due to the low solidification point. Moreover, this point can be lowered by adding, for instance, appropriate salts. Finally the purchase price as well as the depot costs gives nitric acid the preference to peroxide.

Outsiders may really find it unintelligible and discomfoting, that Germany had worked on the basis of three different oxidizers. But this becomes clear if one considers the history of the development. At the beginning of rocket powering with liquid fuels there was only liquid oxygen which seemed to furnish the highest enthalpy grade in fuel mixtures. These early investigations aimed solely at space travel\* and therefore did not consider economic and military interests. The fact that liquid oxygen is nowadays used in modern rocket engines represents a clear atavism.

Peroxide was chosen for the development of submarine engines, because it allowed absolutely traceless under-water travel. The industries furnishing this material for experiments on peroxide turbines tried to secure an expansion of the market, which led to the use of peroxide in rocket engines.

In Germany it was only BMW which, without any atavistic influence and without any direct or indirect scope considerations with producers of oxidizers, was led by economic and military thoughts to the choice of oxidizers and due to their sense of responsibility, decided logically on commercial concentrated nitric acid.

\* One of the greatest rocket pioneers told me once (extracted): "Space travel would not have been possible without the preliminary creditable performances of several great men such as C. von Linde and his air liquefaction system."

# HISTORY OF THE AIR-BORNE TOWING EXPERIMENTS WITH LARGE SIZE RAM-JET DUCTS IN GERMANY DURING 1941 - 1945

IRENE SÄNGER-BREDT \*

## 1. REASONS FOR THE EXPERIMENTS AND REVIEW OF THE STATE OF TECHNICAL DEVELOPMENT OF RAM-JET POWER PLANTS AT THE BEGINNING OF THE EXPERIMENTS

Two short notices constitute the beginning of our ram-jet experiments. The first was published as early as 1913 in the magazine "L'Aérophile" and dealt with a suggestion made by the French engineer RENÉ LORIN for a hypothetical flying body which he himself described as a "propulseur par réaction directe". The text and a drawing accompanying it stated only that a heated aeroduct consisting of an inlet diffuser able to ram incoming air, and attached combustion chamber with suitably shaped expansion nozzle, would show, in a wind tunnel test, not resistance, but thrust.

The other notice came from the German Air Ministry in 1941. It addressed only a very restricted group of persons and called for the design of an anti-aircraft weapon, able to climb up to 12 km in 2 min and to operate at this altitude for almost one hour.

The idea of coupling these two notices was not as evident at that time as it might seem today. Pure ram-jet propulsion was, it is true, at that time not quite forgotten, but had fallen into some discredit because of the experience of those few persons who had either re-invented it or who, starting from LORIN's fragmentary notes, had tried to handle it. Curiously enough, there was nobody after LORIN who would acknowledge the outstanding advantages of the stationary ram-jet engine, namely, the drastic simplicity of its construction, its robustness and its insensitivity as regards the type of fuel. Instead, there was a tendency to try to complicate the engine, to ask too much of it in the most unsuitable applications imaginable, or to weaken the basic concept of the ram-jet propulsion either by combining with it as various as possible propulsion systems at once or (in attempting to improve the aerodynamic efficiency by a few percent) by using special shapes or internal installations which usually interfered considerably with the combustion and therefore necessitated further devices for the maintenance of flame stability. An almost tragic example of

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such tendencies is associated with the name of LEDUC, who is distinguished by having been the first to realize the theory of LORIN in 1936 by results of tests, but who did not succeed in turning out a flying device fit for production despite twenty years of experimenting.

In Germany systematic testing of ram-jet engines on appropriate test stands was started as early as 1937 by HELLMUTH WALTER in Kiel. His engines, which were characterized by low combustion chamber temperatures, not exceeding  $1300^{\circ}\text{K}$ , obtained thrust coefficients of 0.1 at most and this although all troubles of mixing and evaporation were avoided by the use of natural gas instead of liquid fuel.

Owing to the lack of testing experience and of accurate flow- and performance calculations, a triple objection prevailed at this time against the ram-jet system which brought all experiments to a deadlock. It was believed that:

1. the highest thrust coefficients ( $C_T = T/q S_2$ ) were obtainable at most complete ram in the diffuser,
2. combustion temperatures exceeding  $1300^{\circ}\text{K}$  could not be controlled in a ram-jet duct without artificial cooling,
3. the period of time for the flow in the duct would be insufficient to allow satisfactory fuel cracking and fuel-air premixing, and, therefore together with the low gas densities, would result in incomplete, possibly unstable combustion.

Despite these handicaps EUGEN SANGER ventured in 1941 to meet the tactical requirements of the German Air Ministry with the idea of what he called a "modified LORIN engine" which, by admitting very high combustion temperatures, would considerably increase the thrust concentration at the expense of efficiency, thus giving the engine in the high subsonic range a superiority over *rockets* because of considerably lower fuel consumption, and over *turbo jets* because of higher possible values of absolute thrust. He was guided by the following considerations: the ram-jet engine, with its simple construction and its peculiar thrust characteristic, which — similar to air resistance — rises nearly proportionally to the dynamic pressure and to the main cross-sectional area, has, besides its well-known disadvantages (e.g. above all, its zero static thrust), several very definite and typical advantages, which determine its applications and its importance relative to other propulsion systems. These are:

1. Above a given minimum flight speed it is possible to obtain higher thrusts than those of turbo-jet engines, still in the subsonic range, and at the same time a better efficiency coefficient and a lower fuel consumption relative to rockets of the same power.
2. The rate of climb ability of a ram-jet aircraft is about the same as that of a rocket fighter.
3. As opposed to rockets and turbo-jets, the ram-jet engine has a decidedly more favourable throttling behaviour with simultaneously increasing efficiency.
4. It has a smaller power-weight ratio, namely  $1/5$  or  $1/10$  of that of the corresponding turbo-jet engines.
5. Due to the uncomplicated shape of the engine and the absence of moving parts, its construction is inexpensive and it is suitable for mass production,

needs little maintenance, and can be conceived for any power-output simply by varying the cross-sectional area.

6. The engine is largely insensitive to the kind of fuel used.

## 2. THEORETICAL PRELIMINARY WORK AND TOWING TESTS ON TRUCKS

The planning of our work was governed by the desire to obtain results as positively and as quickly as possible: we could not afford time consuming, novel test stand facilities with their associated risks.

The testing of model-size ram-jet engines in the available wind tunnels was impracticable, because the similarity laws of mechanics can only be applied to cold and not to heated engines. Moreover, at that time there existed no wind tunnel which allowed the reproduction of high altitude conditions, and just combustion at lower air densities was one of the main points of our research programme.

Combustion tests in the free air jet of a blower were also expected to be unsatisfactory, due to the difficulty of reproducing at the diffuser intake the conditions (i. e. turbulence, density and humidity), met in an undisturbed free flow.

Up to that time, however, nobody had tested the novel engine in flight, and nobody had yet carried out rigid calculations to find the correct theoretical area ratios between diffuser and nozzle with respect to a given flight condition, ratios on which the performance of the engine depends so critically. We decided to attack the problem on these two points.

The mere computing could be started right away. Its essential results are detailed in Fig. 1 which shows the thrust coefficient and the appropriate nozzle opening ratio for various fuel-air mixture rates and a given flight condition, as a function of the diffuser opening ratio. A similar diagram, scheduled for flight at sea level and a lower MACH number, in its time made it possible not only to give a simple explanation of the low thrust coefficients obtained by WALTER, but also to choose advantageous diffuser opening ratios for our own experiments.

Diagrams of the type of Fig. 1 show thrust coefficients which, beginning at zero corresponding to total ram in the diffuser, increase almost linearly with increasing diffuser opening ratio, until they bend over into a curve with a maximum value for  $C_T$ , the position of which moves with sinking mixture ratio, decreasing flight altitude and increasing MACH number, towards increasing diffuser opening ratios. A little beyond this maximum on the  $x$ -co-ordinate we find at first the point corresponding to flow conditions where a convergent expansion nozzle becomes superfluous, and only with still slightly greater diffuser opening ratios we met the case where, with an already diverging nozzle, thermal choking (i. e. actually unstable flow) occurs. Since the aerodynamic drag coefficients, which have to be deducted from the thrust coefficient to find the net thrust available for acceleration, also have finite values for a diffuser opening ratio of zero, there exists a minimum opening ratio  $S_1/S_2$  for the diffuser, above which a ram-jet duct begins to be of technical interest. On the other hand, the velocity of the fresh air, chosen as low as possible at the

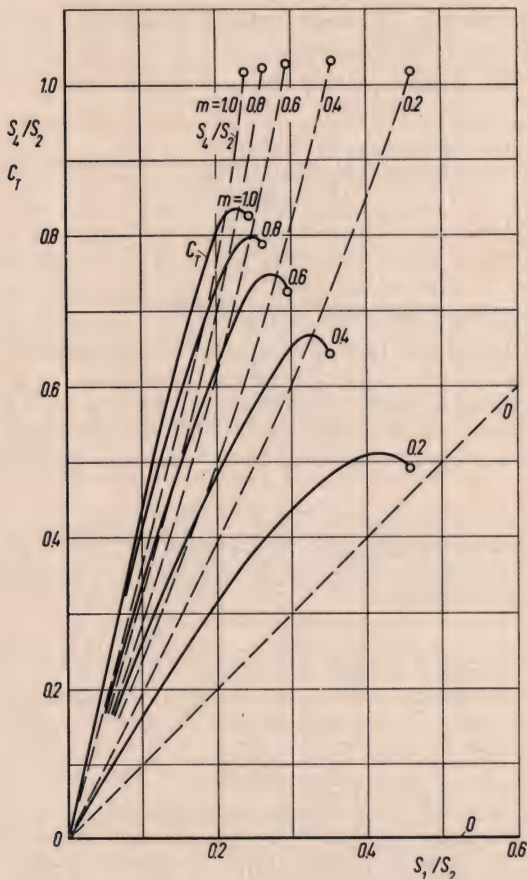


Fig. 1. Theoretical values of the thrust coefficient  $C_T$  and the nozzle opening ratio  $S_4/S_2$  as a function of a chosen diffuser opening ratio  $S_1/S_2$  assuming one-dimensional isentropic flow as well as ambient pressure in  $S_1$  and  $S_4$ , for the following flight conditions: 11 km altitude, flight Mach number 1 and several selected values for fuel/air mixture rate  $m$

diffuser outlet with regard to preparation of mixture and combustion processes, increases almost linearly with the diffuser opening ratio. Consequently, for our first experiments, we decided to choose as a cautious compromise the ratio of  $S_1/S_2 = 0.16$  — a figure lying still on this side of the maximum for  $C_T$ . As far as we could learn from available records on WALTER's tests, he had worked up to that time with diffuser opening ratios varying from 0.02 to 0.11 and therefore could not, even if he had worked with flows free of any losses, obtain thrust coefficients higher than about 0.4. It was this knowledge which at that time increased to a certain extent our still timid confidence in the ram-jet system.

Before we ventured to approach proper air borne tests, we tried to clear up a number of critical problems beforehand by ground-towing tests. Above all, it seemed important to find out whether the high combustion gas temperatures

imposed could be realized, whether combustion remained stable, and whether the cooling by air-stream alone would suffice to protect the parts of duct surface exposed to combustion gases, even in the case of low driving speed and of stationary high temperature combustion. In addition to this, measurements of the static pressure along the diffuser inside should decide the optimum diffuser inlet angle which would result as a compromise of a still reasonable diffuser efficiency and a tolerable diffuser length.

Some of these problems could be solved by towing tests with *combustion chambers only*; here the towing speed corresponded to the assumed air-speed at the diffuser outlet. Other tests required the towing of the *complete ram-duct*. An OPEL-BLITZ truck was used a towing vehicle. The speed attained with a mounted jet duct was, for that time, remarkably high and reached 90 km/h on the test course. The test duration was limited by the length of the road-way to only a few minutes. Our first test runs began in early autumn 1941 with a violation of sacred red-tapism of our administration. As unsuspecting idealists we imagined we would earn particular merits in those critical periods by doing our research work as quickly and economically as possible. Thus, our enthusiasm led us to misuse a simple drain pipe of approximately 50 cm diameter and several metres in length, belonging to our construction department. After we had rather proudly transmitted our first good test results to Berlin, to our stupefaction we earned instead of expected commendations only a cool reprimand, criticizing the fact that we dared to produce successful test results before the tests themselves were officially authorized. Nevertheless, somewhat later we continued our testing with official authorization using properly constructed combustion ducts of 800 mm diameter as well as complete ram-jets with a maximum diameter of 500 mm and with interchangeable diffuser. Such a test run is shown in Fig. 2.



Fig. 2. Road towing test on 27th October 1941 with high-temperature combustion chamber of 800 mm diameter

The towing tests with *combustion chamber only* led to the design of an injector system to be constructed mainly as a grate consisting of widely spaced tubes carrying atomizer nozzles with spin effect and ejecting the fuel under a pressure of 5 to 10 at against the direction of motion, thus giving the best atomization. The ignition was first accomplished by the use of diethyl of zinc; later on spark plugs were used. In addition, combustion temperatures of more than 2200 °K were measured by observing melting points of thin wires of rare metals, such as, for instance, platinum-iridium, etc. In spite of these stationary temperatures, no effect on the combustion chamber walls could be observed.

The road towing tests with *complete duct* resulted in diffuser efficiencies of more than 90% for divergence angles below 8° and of about 85% for 10° angles. Still larger angles led to an abrupt loss of efficiency down to the theoretical figures obtained by assuming the BORDA-CARNOT pressure loss.

The towing tests were not only run with gasoline, but also with gas-oil and with aluminium dispersed in gas-oil. The latter gave difficulties as far as pumping and atomizing were concerned.

The results of this first testing phase were set down in the German Patent "DP 165 144" in 1941 under the title "High Temperature Ram-Jet Engines" (Hochsttemperatur-Staustahltriebwerke).

### 3. AIR-BORNE TOWING TESTS WITH 500 mm DUCTS ON DORNIER AIRCRAFT Do 17 Z

The preliminary theoretical work and the results of the ground towing tests led to a series of principles for the design of the ram-jet test ducts which proved to need no alteration in the years following. These systematic features were:

axially symmetric ducts of varying cross sectional area, with uncooled steel walls of thickness 1 to 4 mm, consisting of 4 construction elements: the diffuser, combustion chamber, exit nozzle, and — as the only internal device — the injector grid.

In the construction of the diffuser the following classical rules were recognized: — circular flow area, pure conical diffuser shape, smooth and curved transition from intake to diffuser cone, utmost smoothness of the inner surfaces, small divergence angles, etc. The use of a pure inlet diffuser was waived in spite of the interesting efficiencies in the subsonic range, because this better efficiency was only to be expected within a small action range and was to the detriment of efficiencies in a larger performance range. Moreover, we intended already at that time, when the maximum flight speed realized by manned aircraft was only about  $M = 0.8$ , to enlarge the application possibilities of the engine into the supersonic range by admitting a vertical shock-wave. Fig. 3 illustrates this idea, showing the overall efficiency — as a function of flight MACH number — of a supersonic ram-jet engine with vertical shock diffuser and compared with other types of engines.

The combustion chambers had a cylindrical form and originally were shaped longer than presumably needed, to allow for reduction according to the results of the combustion efficiency measurements. The exhaust nozzles were cylindrical truncated cones like the diffusers; for test flights they were not

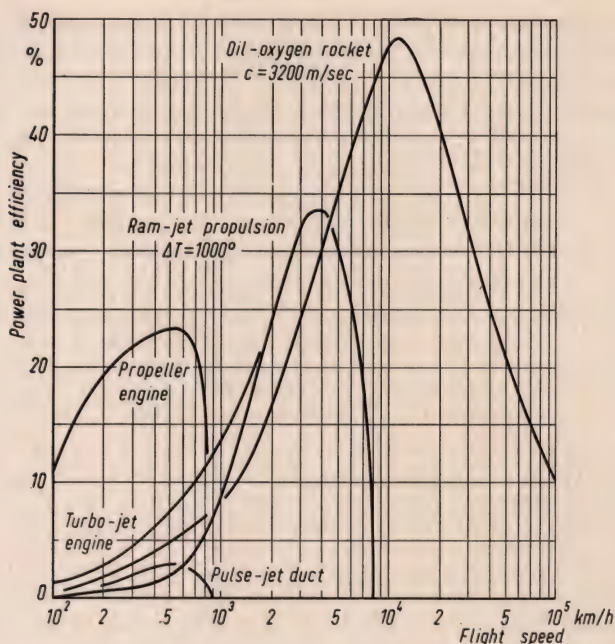


Fig. 3. Overall efficiency of different power plant systems versus flight Mach number

provided with adjustment devices, but were interchangeable in order to allow opening ratios according to the theoretical values for the different flow conditions under consideration.

The combustion chamber diameter used in the first test series, which comprised 70 flights, was 500 mm, the length of the chamber 4 m and the diffuser cone angle  $10^\circ$ . Most of the tests were flown with a diffuser opening ratio  $S_1/S_2 = 0.158$  and with nozzle opening ratios varying between 0.400 and 0.640. Several tests with  $S_1/S_2 = 0.236$ , i. e. with increased cold air speed and a duct without nozzle, at which an increase of thrust was expected theoretically, did not show, it is true, exactly a noticeable slope of the thrust coefficient, but resulted in a less stable combustion.

Apart from the varying of proper duct dimensions, testing went on with variations within the injection system, as for instance variations of injection direction and injection pressure as well as of size, number and distribution of the injection nozzles. Also, the effects of additional construction elements in the diffuser were investigated.

The operating conditions of the test flights varied as follows: almost constant flight MACH number of 0.3; altitudes between 100 and 3000 m; fuel-air ratios between 80 and 140% of the stoichiometric mixture. The test fuels were gasoline (with octane numbers between 40 and 87) and methanol.

The testing time of these flights was limited by the capacity of the supplementary tanks (81 litres) and lasted about 5 min in the case of stoichiometric mixture ratios, and adequately longer when the combustion was throttled.

Besides the standard test conditions in flight, such as dynamic pressure, static pressure and temperature of the surrounding atmosphere, injection pressure, combustion time, etc., on every flight the measuring was extended to the pressure excess over atmospheric pressure in the flow inlet and outlet area of the combustion chamber. The figures here obtained allowed, by comparison with the adequate theoretical values, the computation of diffuser and combustion efficiencies as well as, by using a general formula for incompressible flow, the computation of thrust coefficients. Furthermore, the free thrust exerted on the carrier plane by the ram-jet in operation was measured by a dynamometer. This allowed the computation of the coefficient of the net thrust of the ram-jet itself, provided that cold and warm drag coefficients were known. The cold drag coefficients were found by separate test flights and, in addition to this, by special model tests in wind tunnels. Moreover, the interesting drag coefficients of the heated ram-jet were computed by assuming most unfavourable conditions. Interference drags, which could occur, were more or less neglected. In spite of this rough procedure the agreement between the thrust coefficients calculated this way and those obtained by measurement of pressure variation was satisfactory. Finally, the top temperatures of the combustion chamber walls were measured thermocolourimetrically.

The first air-borne towing test with a ram-jet engine was carried out on 6th March 1942. Fig. 4 shows one of the first flights at about 2000 m altitude with stoichiometric gasoline combustion. This picture also shows clearly the way the ram-jet engine was attached to the carrier plane during the first test series. Later on, when dynamometer readings were no longer needed, this system was replaced by an aerodynamically smoother one.



Fig. 4. Test ram-jet of 500 mm cross-sectional diameter operating at a stoichiometric fuel/air ratio and towed on Do 17 Z, with flight speed of 87 m/sec

194 MPA

Table 1 shows the results of some of the most successful test flights. Their main conclusions are:

1. Good agreement between measured and computed thrusts and fuel consumptions. — The measured thrust coefficients varied between 0.44 and 0.58, according to operation conditions, and under the best test conditions reached 94% of the computed loss-free figures. The type of fuel had no perceptible influence on the thrust coefficients, but it did affect fuel consumption, and here again it was highest in the case of methanol due to the implicit oxygen load.
2. The noise of the engine was moderate and similar to a steady roar; combustion remained steady except in the cases of very excessive fuel supply

and of far too narrow nozzle exit areas. The normal form of the exhaust flame is best seen in Fig. 4. In the case of methanol combustion, the flame was absolutely colourless.

3. As far as the injection system is concerned, the experiences based on the road towing tests were confirmed and the great influence of high injection pressures on the combustion efficiency became evident. The thrust coefficients measured were, for a constant number of injection nozzles, highest when the injection pressure was at its highest value of about 36 at. At a reduced injection pressure of 4 at, they decreased to about 80% of the optimum.
4. The best diffuser efficiency obtained was 90%. Streamlined section struts mounted inside the diffuser approximately  $\frac{2}{3}$  from its intake end had less influence than minor reductions of the injection pressure.
5. The highest temperatures of the combustion chamber walls measured thermo-colourimetrically were found to be between 500 and 600 °C.

#### 4. AIR-BORNE TOWING TESTS ON INCLINED FLIGHT PATH WITH 1500 mm DIAMETER DUCTS ON Do 217 AIRCRAFT

Through the good offices of our very keen and interested test pilot PAUL SPREMBERG it was possible to use a Do 217 E2 bomber for further tests. This aircraft proved to be an excellent carrier after several insignificant alterations had been made. We wanted to test with it the functioning of larger ram-jets at maximum subsonic speeds.

The new test duct had a diffuser intake diameter of 600 mm and a cone angle of 10°, a combustion chamber 4000 mm long and 1500 mm in diameter, i. e. a rated power of about 20,000 HP at a flight speed of 300 m/sec. The nozzle opening ratio was  $S_4/S_2 = 0.640$ . Thus, the test object had a total length of 10,600 mm and weighed about 2000 kg including pump system and test fuel load.

As shown in Fig. 5, which gives an overall view of the carrier with mounted ram-jet, even a powerful craft like the Do 217 was seriously affected by such a mighty arrangement. Stability and centre of gravity were considerably disturbed. In spite of careful computing and model testing in the DORNIER wind tunnel, the testing of the flight characteristics themselves could still provide surprises.



*Fig. 5. Side view of the test carrier Do 217 with mounted 1500 mm duct*

Table 1. Results of air-borne tests with high temperature ram-jet engines of

1	2	3	4	5	6	7	8
Test Nr.	Date	$H$ [m]	$v$ [m/sec]	$\alpha$ [°]	F. T.	I. N. $\phi$ [mm]	I. D.
41	27. 5. 42	500	80.2	2.8	a	6 $\phi$ 1.65; 1 $\phi$ 2.7	d
43	3. 6. 42	1950	88.2	2.0	a	6 $\phi$ 1.65; 1 $\phi$ 2.7	d
44	3. 6. 42	1200	86.3	2.0	a	6 $\phi$ 1.65; 1 $\phi$ 2.7	d
45	4. 6. 42	1900	82.6	2.8	a	7 $\phi$ 1.65	e
48	4. 6. 42	2000	86.9	3.0	a	7 $\phi$ 1.65	f
49	6. 6. 42	550	81.0	2.4	a	7 $\phi$ 1.65	f
50	6. 6. 42	850	82.9	2.1	a	7 $\phi$ 1.65	f
51	6. 6. 42	850	82.9	2.6	a	6 $\phi$ 1.65; 1 $\phi$ 2.7	f
52	6. 6. 42	1200	86.3	2.5	a	6 $\phi$ 1.65; 1 $\phi$ 2.7	f
53	8. 6. 42	1000	84.3	2.4	a	6 $\phi$ 1.65; 1 $\phi$ 2.7	f
55	8. 6. 42	1000	82.6	2.6	a	7 $\phi$ 1.65	g
56	8. 6. 42	1000	86.4	1.3	a	6 $\phi$ 1.65; 1 $\phi$ 2.7	g
57	8. 6. 42	1000	80.7	2.8	a	7 $\phi$ 1.65	g
58	9. 6. 42	1500	83.8	2.2	a	12 $\phi$ 1.65	h
59	9. 6. 42	1300	84.0	1.8	a	12 $\phi$ 1.65	i
61	11. 6. 42	400	81.6	2.3	a	14 $\phi$ 1.65	i
62	11. 6. 42	2000	85.8	2.3	a	14 $\phi$ 1.65	i
63	11. 6. 42	2000	86.5	2.1	a	7 $\phi$ 1.65	i
64	11. 6. 42	2000	87.5	2.2	a	7 $\phi$ 1.65	i
65	11. 6. 42	2000	85.3	2.3	a	7 $\phi$ 1.65	i
66	11. 6. 42	2000	87.5	2.2	a	7 $\phi$ 1.65	i
67	12. 6. 42	1000	84.1	1.8	b	14 $\phi$ 1.65	i
68	12. 6. 42	1200	85.5	1.6	b	14 $\phi$ 1.65	i
69	12. 6. 42	1200	83.5	1.8	b	7 $\phi$ 2.7	i
70	12. 4. 42	100	78.5	2.5	c	14 $\phi$ 1.65	i

 $H$  Flight altitude $v$  Flight speed $\alpha$  Angle of incidence

F. T. Fuel type

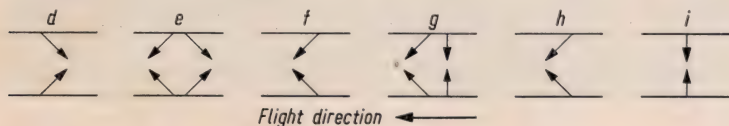
a) Gasoline of octane value 40; density 0.715 kg/dm<sup>3</sup>

b) Aviation gasoline B 4 of octane value 87

c) Methanol

I. N. Number and size of the injector nozzles

I. D. Injector direction



I. Sanger-Bredt, Air-Borne Towing Experiments with Ram-Jet Ducts

500 mm combustion chamber diameter

9	10	11	12	13	14	15	16
$p$ [at]	$B$ [kg/sec]	$m$	$\Delta$ [kg]	$\Delta p_2/q$	$\Delta p_3/q$	$C_T^*$	$C_T^{**}$
13	0.225	1.16	52.7	0.66	0.21	0.44	0.44
11	0.220	1.19	51.2	0.66	0.23	0.40	0.43
11	0.233	1.20	51.0	0.66	0.23	0.38	0.43
10	0.181	1.04	52.3	0.72	0.25	0.52	0.47
32	0.215	1.19	57.6	0.80	0.27	0.51	0.52
32	0.235	1.20	60.3	0.71	0.29	0.53	0.46
32	0.232	1.20	60.8	0.73	0.29	0.50	0.47
11	0.186	0.96	51.2	0.64	0.26	0.41	0.41
14	0.213	1.09	54.4	0.68	0.27	0.46	0.44
15	0.222	1.15	58.6	0.70	0.29	0.47	0.45
36	0.257	1.35	62.8	0.77	0.28	0.56	0.57
16	0.249	1.25	62.0	0.88	0.32	0.48	0.53
36	0.263	1.41	60.2	0.81	0.27	0.56	0.57
7 <sup>1/4</sup>	0.215	1.17	56.9	0.73	0.29	0.49	0.47
7 <sup>1/4</sup>	0.216	1.15	53.0	0.73	0.29	0.42	0.47
22	0.232	1.17	57.4	0.75	0.22	0.46	0.50
22	0.214	1.20	55.1	0.71	0.22	0.48	0.47
34	0.270	1.50	63.8	0.86	0.31	0.58	0.56
34	0.254	1.39	65.4	0.86	0.30	0.58	0.56
34	0.252	1.42	60.5	0.83	0.29	0.56	0.55
34	0.249	1.36	63.6	0.83	0.29	0.56	0.54
8	0.206	1.06	60.8	0.75	0.27	0.50	0.49
12	0.245	1.27	62.2	0.81	0.27	0.51	0.53
4	0.225	1.19	54.9	0.67	0.20	0.45	0.44
36	0.50	1.08	59.7	0.85	0.28	0.53	0.56

$p$  Injector pressure

$B$  Fuel supply

$m$  Fuel/air ratio as a multiple of the stoichiometric ratio

$\Delta$  Dynamometer indication ( $\Delta = T_{\text{heated}} - T_{\text{cold}}$ )

$\Delta p_2/q$  Ratio of the measured excess pressure in the combustion chamber inlet section to the conventional stagnation pressure

$\Delta p_3/q$  Ratio of the measured excess pressure in the combustion chamber exit section to the conventional stagnation pressure

$C_T^*$  Thrust coefficient evaluated from the dynamometer measurement

$C_T^{**}$  Thrust coefficient evaluated from the pressure measurements in  $S_2$  and  $S_3$

Indeed, certain warning voices, directed at keeping flight captain SPREMBERG and his crew back from such dangerous tests, were not missed. Actually, the mechanic resigned, but since Dr. SANGER, who was in charge of the experiments, insisted on participating on board himself in the new flight test series as he had done in the earlier series, another mechanic was soon encouraged. For the first take-off the air-base commander sent fire wagons, ambulances and so much other emergency equipment that the pilot was of the opinion that only the hearse had been forgotten and the stomachs of the ground observers turned over. Fortunately, however, all the flights ended happily and without serious incidents.

The object of these tests was mainly to give a rough idea of the thrust and the fuel consumption of the large duct at speeds from 100 to 200 m/sec and to watch the combustion under these conditions. The mounting of the duct is shown in Fig. 5. In order to permit attachment of the duct to the carrier, a streamlined strut leads through the clear part of the diffuser at about  $\frac{2}{3}$  of its length. The fuel was forced by pressurised nitrogen from a 700 litres pressure tank into the fuel pipe and ejected at the end of the diffuser through 60 to 120 atomizer nozzles into the combustion chamber with a pressure ranging between 4 and 11 at. The injector system and the nozzle arrangement remained the same as in the earlier tests.

During the tests, multi-tube manometers measured the excess pressures in the specially designed cross-sections, such as the diffuser inlet and combustion chamber inlet and exit, as well as the dynamic pressure of the surrounding air and the injection pressure. These manometers were mounted in the cabin of the carrier, and the results registered together with the testing time by a robot camera. Since the carrier with mounted duct achieved a horizontal speed of only 90 m/sec with its own engines and a speed of approximately 110 m/sec with the ram-jet engine burning, higher speeds could only be obtained by powered glide of the carrier aircraft. Therefore, the speed range from 100 to 200 m/sec was always flown in an accelerating glide. Fig. 6 shows the carrier plane with burning ram-jet at maximum speed. The snap was taken from ground by tele-objective.

Due to the small variability of the fuel supply by the injection pressure, the fuel excess decreases with rising flight speed. Therefore each test flight



*Fig. 6. Test plane Do 217 with operating ram-jet engine at a flight speed of 200 m/sec*

furnishes a series of measurements at various flight speeds and at corresponding fuel-air ratios. The fixed cross-sectional dimensions of the duct, however, are rigorously appropriate for only one of the operating conditions in this range; all the others result therefore in increased efficiency losses. The number of flights in these tests was 18.

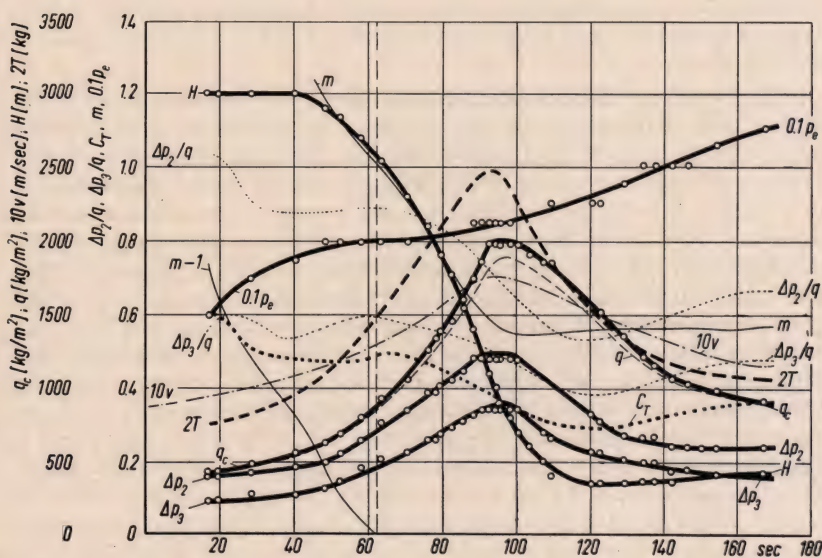
Table 2 gives the results of some of the most revealing test flights. Its circles correspond to observed positions of the direct measurements, as flight altitude, dynamic pressure of flight, injection pressure and static excess pressures inside the duct; the resulting values for flight speed, fuel-air ratio, thrust coefficient, etc. are given only by curves. They fully confirm the results of theoretical computation and of the test flights with smaller ducts. The combustion grew steadier with increasing flight speed; it was on the whole considerably steadier than in small ducts and remained positively within the duct at each fuel-air ratio. In spite of the fact that the injection pressures were too low throughout, the diffuser efficiencies at combustions with stoichiometric or overstoichiometric fuel-air ratios are in part remarkably high and, moreover, show a tendency to increase even more with rising injection pressure. Stoichiometric combustion results in thrust coefficients of between 0.43 and 0.50; overstoichiometric fuel injection increases this to 0.60. In this connection, it should be mentioned that the estimated values of  $C_T$  of 0.43 to 0.50 may be the result of understoichiometric combustion already due to a certain inertia of the board instruments. New in face of Table 1 are measurement results of test flight periods with substoichiometric fuel supply and with consequently too large nozzle exit cross-section areas. In spite of these unfitted area sizes the measurements show still satisfying results within the range of tested flight speeds. However, at maximum flight speeds the ram-jet could in any case only operate with substoichiometric fuel ratio, because otherwise, according to the pilot's report, the horizontal stabilizers were no longer able to compensate the nose heaviness of the carrier, produced by the ram-jet thrust, and the aeroplane started to become unstable. — Measurements of the surface temperatures of the duct by a thermocolourimetric paint gave no new results in comparison with previous experiences.

## 5. PRECISION MEASUREMENTS ON A 1000 mm DUCT ON CARRIER Do 217

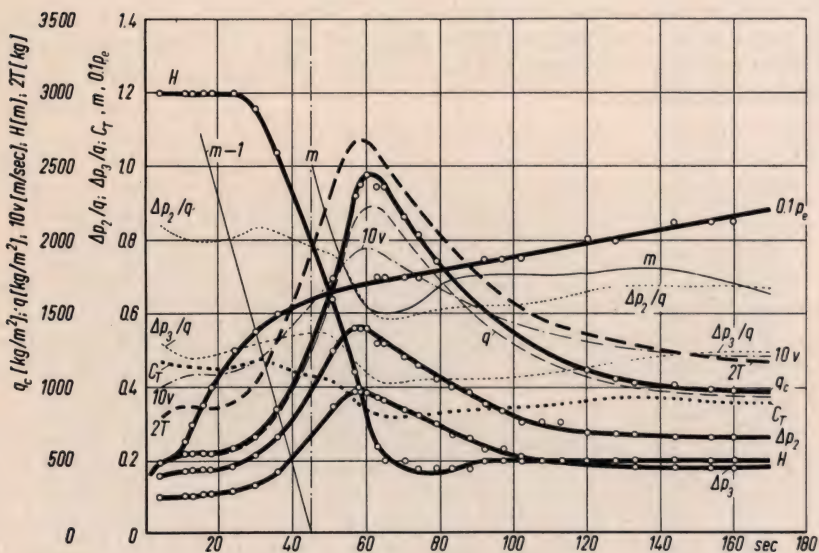
Whereas the air-borne towing tests hitherto described were of a rather general and instructional nature and mainly designed to furnish qualitative information, new flight tests with a 1000 mm duct were to allow more exact measuring, using as a basis the experience gained with the 500 mm and the 1500 mm ducts.

Since the performance of larger sized ducts had already been investigated on principle, the size of the main cross-section of the engine was chosen with regard to its supposed application as an additional power plant, as a missile engine, or in a small sized fighter. The diffuser was made of simple 3 mm sheet steel, the fire-exposed wall parts were made of 4 mm steel Sicromal 3. The parts were assembled by welding. The excess control pressure exerted on the internal surfaces of the whole duct was 30 at.

Table 2. Results of air-borne towing tests with high temperature duct of

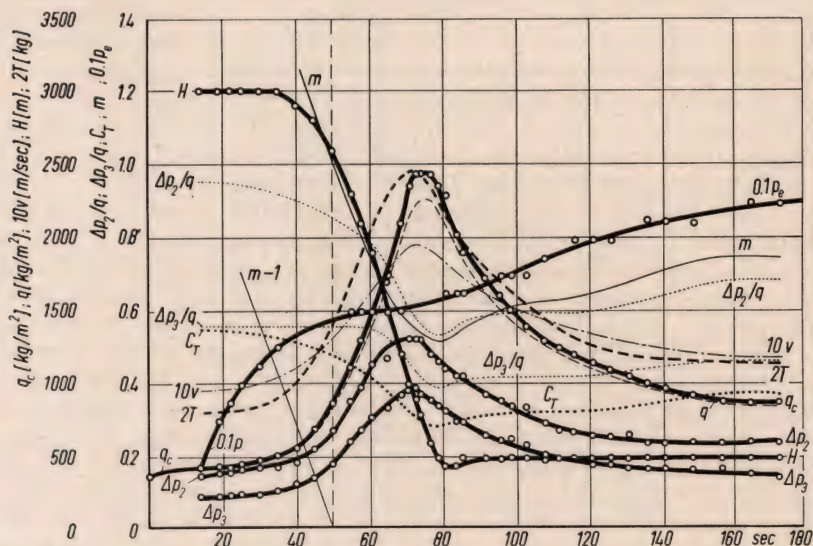


a) Test No. 84, 12th January 1943. Maximum flight speed 177 m/sec. 82 Richter nozzles, 2.7 mm diameter at 10 at.

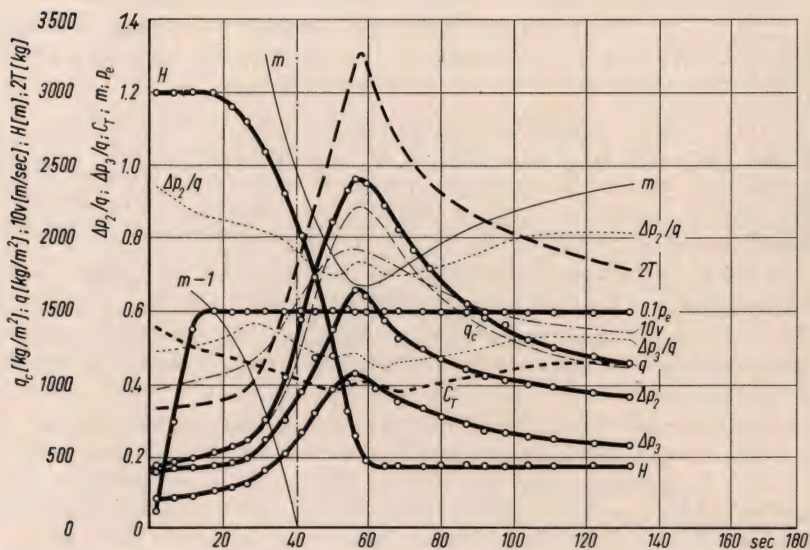


b) Test No. 86, 18th January 1943. Maximum flight speed 196 m/sec. 82 Richter nozzles, 2.7 mm diameter at 10 at.

1500 mm combustion chamber diameter. ○ Measured values.



c) Test No. 87, 18th January 1943. Maximum flight speed 197 m/sec. 119 Richter nozzles, 2.7 mm at 10 at.



d) Test No. 88, 20th January 1943. Maximum flight speed 194 m/sec. 82 Richter nozzles, 2.7 mm diameter at 10 at.

Here again the diffuser was built as an intake diffuser with the front edge only slightly curved, a cone-angle of  $10^\circ$  and an opening area ratio  $S_1/S_2 = 0.158$ . For measuring purposes the length of the cylindrical combustion chamber was again chosen as 4000 mm. Like the diffuser, the form of the nozzle was a truncated cone, but with an area ratio of 0.565 and a cone-angle of  $11.75^\circ$ . This means that the size of the nozzle exit area was designed to give the cylindricform undisturbed air stream — entering the diffuser in the case of stoichiometric combustion and in moderate flight altitudes — a base equal to the area of the diffuser throat  $S_1$ . With the exception of a stiffening cross strut which was part of the attachment of the duct to the carrier, the only internal part was the fuel injector grid, constructed as follows: a cross of 21 mm steel pipe was located 25 mm behind the welded seam between diffuser and combustion chamber and was interspersed by a circular ring of 18 mm steel pipe. The ring had 28 equiaxed thread connections located 75 mm from one another, and the cross itself had another 12 connections, giving a standard total of 40 atomizer nozzles on the injector. When all of them were not needed, those that were not being used were plugged; when more were needed, twin nozzles had to replace some of the normal nozzles. The type of injection nozzle used was again the swirl nozzle. As fuel, only aviation gasoline B 4 was used and this was forced by gas under a pressure in excess of 20 at from a pressure tank of 0.6 m<sup>3</sup> capacity, which replaced the standard body tank, into the fuel pipe system. Ignition was obtained by an electric spark plug situated about 80 mm before the injection system in the bottom of the diffuser, and consequently within a space of high turbulence.

The flight programme consisted of:

1. test flights at full power and at altitudes between 1 and 4 km,
  2. full power flights up to 7 km altitude to test altitude behaviour,
  3. throttled and non-powered flights at altitudes between 1 and 7 km.
- All measurements were carried out under fully stationary operating conditions.

The internal and exterior forces acting on the duct were measured by the following methods:

The pure thrust resulting from the excess pressure on the inner surfaces of the duct was computed from numerous pressure readings along the inside of the duct wall by integration of the pressure components in the direction of flight. Fig. 7 shows the pressure distribution measured at stoichiometric combustion and this is compared to the theoretical loss-free pressure distribution.

The exterior forces on the duct between front and back ram point, i. e. the form drag and skin friction, were determined by measurements of impulse loss.

The inner frictional resistance, which later on was measured by cooled impulse rakes, was at that time theoretically determined by assuming the most unfavourable conditions.

Besides the static pressure and dynamic pressure measurements along the inner surfaces and within the exterior flow around the duct, the ram pressure and static pressure distributions at the diffuser throat cross-section were measured as well, as those at the nozzle exit, the latter only at non-powered flight.

The complete measuring system is illustrated by the sketch in Fig. 8. It shows the location of the 26 pressure measuring points within the

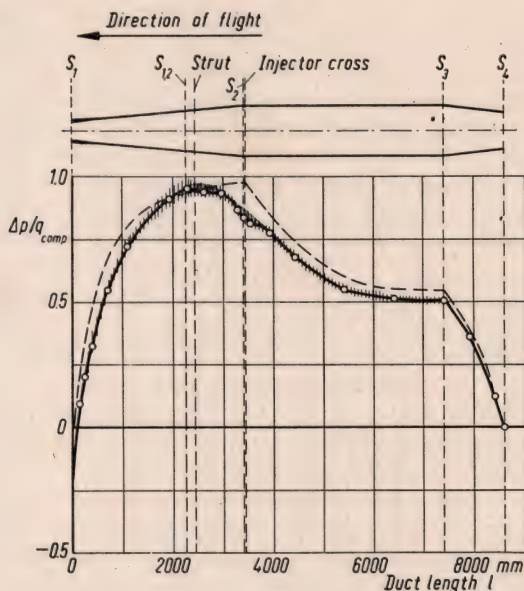


Fig. 7. Comparison of measured (full line) and calculated (dotted line) pressure distributions along the inner duct surfaces at full powered test flight in 4 km altitude. (Hatched area: space of medium error)

duct, and gives the dimensions and position of the impulse measuring rake at the nozzle exit edge. Each pressure measuring point was connected to the multi-tube manometer in the measuring and registration chamber by an aluminium pipe. The chamber was suspended by rubber straps and thus safeguarded against vibrations, with its direction perpendicular to the flight path to prevent the influences of acceleration. The multi-tube manometer was also connected to the measuring points of the flight dynamic pressure and of the static pressure of the undisturbed atmosphere. The last mentioned pressure served as a basic value for all other pressure measurements. The readings of the multi-tube manometer and the instrument panel were automatically and simultaneously photographed by a robot camera at intervals adjustable between 5 and 60 sec and with an exposure of 0.01 sec. On the instrument panel were located, besides a time interval recorder with seconds for the determination of the relative test time, a fuel pressure gauge and three speed gauges for the control of the readings of flight dynamic pressure and of the static overpressures at the combustion chamber inlet and exit. More details appertaining to measuring arrangements and analysis would exceed the scope of the report but can be found up in a detailed description printed in the VDI-Forschungsheft No. 437.

The total number of test flights of this series with a 1000 mm duct was 32. The flights were made at MACH numbers between 0.30 and 0.37, at altitude between 1 and 7 km and with a fuel supply varying from 0.11 to 1.4 of the stoichiometric fuel-air mixture. Fig. 9 shows the test carrier plane during a test flight with fully powered duct, photographed from an accompanying aircraft.

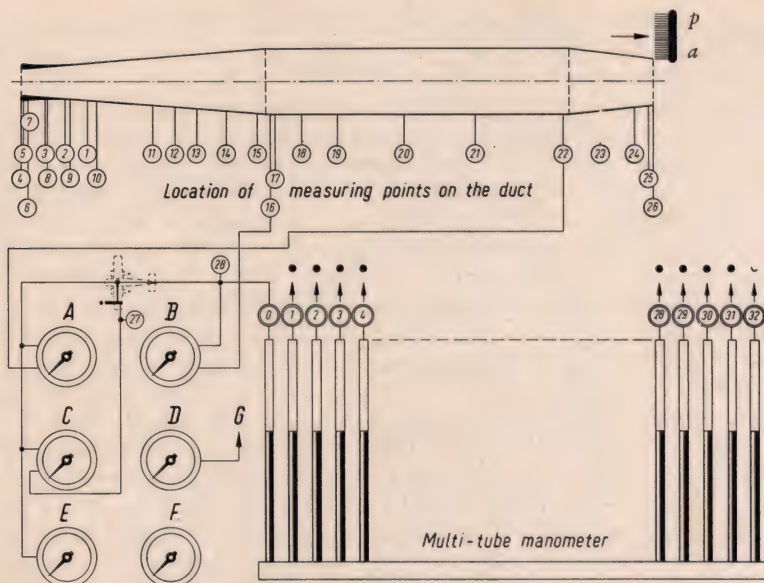


Fig. 8. Measuring system for flight tests with 1000 mm duct

1—4: Pressure measuring points on the exterior wall of the diffuser

5—26: Pressure measuring points along the inside of the duct

For each test, the pressure measuring points 1 to 26 as well as the flight ram pressure point 27 and the basic pressure point 28 are connected to the columns 0 to 32 of the multi-tube manometer as far as needed, by using pressure tubes with capillary damping. The feelers *a* to *p* of the impulse rake are, as far as needed, connected to the columns of the multi-tube manometer.

*A* = Machmeter controlling excess pressure at measuring point 22

*B* = Machmeter controlling excess pressure at measuring point 16

*C* = Machmeter controlling ram pressure measuring

*D* = Injector pressure meter

*E* = Altimeter

*F* = Clock

*G* = To fuel injector pipe



Fig. 9. Do 217 with 1000 mm test ram-jet at full power in 2 km altitude

The full-power tests in moderate flight altitude led to the following technical conclusions: combustion was achieved in the chamber with 100% efficiency. The average thrust coefficient of  $C_T = 0.55 \pm 0.01$  shows losses of about 5% in comparison with the theoretical value. The greater part of these losses may

be due to flow separation, the remainder (about 1%) may be caused by anticipated heating of the flow upstream within the last third of the diffuser in consequence of flame gas radiation and by internal construction devices furnishing resistance. The limiting cross sectional areas can be regarded as suitable for the flow within the bounds of test accuracy. The total of the resistance coefficients measured at the external flow round about the duct varies from 0.07 to 0.09 and depends on whether the combustion is rigorously stoichiometric or overstoichiometric. The calculated coefficient of the inner frictional resistance was 0.03, leaving a coefficient of 0.45 for the free thrust at full-powered operation at moderate altitudes.

The test conditions for 7 km altitude and  $M = 0.35$  for the investigation of altitude influence on combustion already represent, in view of future applications, a rather unfavourable case. The combustion chamber pressure here corresponds, for instance, to an altitude of 10 km in the case of flight MACH number 0.9, or to 18 km altitude in the supersonic range at  $M = 2$ . Consequently, the combustion during these tests was sensibly less steady than in stoichiometric combustion tests at lower altitudes. In spite of this, nearly-stoichiometric combustion tests resulted in thrust coefficients of 0.53, which is 89% of the theoretical value. In this connection it must be remembered that at higher altitudes and at full-powered operation the nozzle exit area of the test duct was too small to permit a smooth flow and, moreover, that the values of the internal pressure distribution measured on the bottom of the duct give a somewhat interpolated, unfavourable value for the thrust in the case of a higher

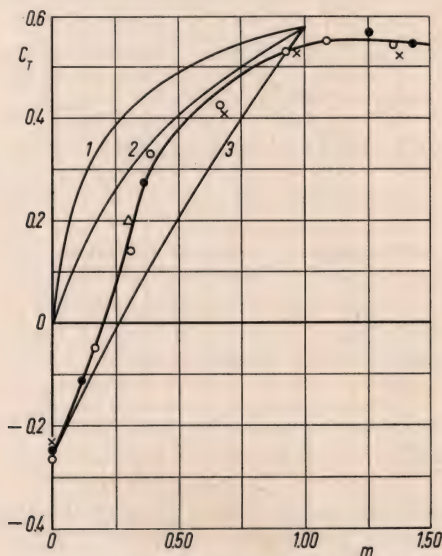


Fig. 10. Gross thrust coefficient  $C_T$  measured as a function of fuellair mixture ratio coefficient  $m$ , and in comparison thereto, analogous theoretical curves for different control systems

1 = Regulation of  $S_1/S_2$  (boost diffuser)    2 = Regulation of  $S_4/S_2$  (adjustable nozzle)  
3 = Regulation by internal resistance bodies

● Test results at 2 km altitude  
○ Test results at 4 km altitude

△ Test results at 3 km altitude  
× Test results at 7 km altitude

angle of incidence of the duct, as happens at higher flight altitudes. The readings of the pressure drop along the combustion chamber indicated that at an altitude of 7 km combustion would close down later than at 4 km, but there would be no difference of efficiency in the case of a duct longer than 3 m. The influence of an inclination angle, of internal construction devices and of other unsymmetrical conditions, is most observable in those cases, in which sound flow conditions exist primarily, as it occurs — with the size chosen for the nozzle exit cross-sectional area — in the case of almost full-power operation. Therefore, it becomes evident that in the case of strongly throttled operation with unchanged nozzle exit area the test results did not vary much from 7 km down to lower altitudes. For the tests flown with strongly throttled operation and without any combustion, the chosen duct exit area proved to be too large, which was turned out by flow separations. Nevertheless, even in these cases, the thrust coefficients obtained were not too unsatisfactory, as is shown in Fig. 10, which compares the variation of the gross thrust coefficient, measured against fuel-air ratio, to the computed curves assuming different control devices, namely perfect inlet diffuser, combustion nozzle adjustment or adjustment by special resistance bodies within the internal flow region of the duct. Fig. 11 shows in an analogous manner the coefficients of the measured thrust after deduction of the internal frictional resistance, as well as the free thrust coefficient which remains after deduction of all internal and external resistances.

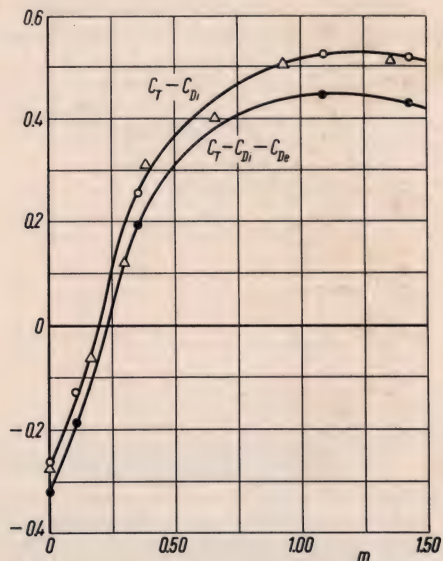


Fig. 11. Coefficient of the measured gross thrust after deduction of internal friction drag ( $C_T - C_{Di}$ ), and free thrust coefficient ( $C_T - C_{Di} - C_{De}$ ), as functions of the fuel/air mixture ratio coefficient  $m$

Originally, it was planned to extend the test programme just described to altitudes up to 15 km and MACH numbers up to 0.8 by towing tests with two 1000 mm ducts on a Me 262 as carrier. The theoretical preparations were already done and had been partly published. The plan failed because of the war situation.

## 6. DETAIL TESTS IN WIND TUNNELS AND BEHIND BLOWERS

The last test flight was performed on 30th August 1944. From then on, the fuel shortage in Germany forced us to restrict our programme to detailed investigations on fixed ground installations.

Besides extensive theoretical investigations, which involved

1. the extension of the thermodynamic computing programme for the dimensioning tables of thermodynamic, constructional and performance properties of ram-jet engines,
2. the flight mechanical performance and optimum investigations,
3. the development of fuel injection devices working with air flow instead of with pressure tanks as earlier,
4. the development of automatically operating fuel regulation devices,
5. the development of regulating devices for the size of the exhaust nozzle,
6. the development of mechanical regulating devices for the injector nozzle system,
7. the designs for installation of ram-jet ducts in existing fuselages or for independent ram-jet aircraft,

(of which the results can best be found in the publications named at the end of this paper) we continued on the following test programme:

1. combustion chamber tests with gas oil behind a blower,
2. combustion chamber tests with lump coal, briquettes and foam coal behind a blower,
3. combustion tests with small hydrocarbon flames in a low pressure chamber,
4. model tests in a smoke tunnel for the study of flow variations due to internal resistance bodies and cross-section variations,
5. model tests in a shallow water channel for the study of supersonic properties of the ram-jet shape.

*The combustion chamber tests with gas oil* were primarily scheduled for the development of a pre-evaporator unit, since the ignition devices in use for the liquid fuels in application up to that time did not work properly with gas oil. Fig. 12 shows such a combustion test with pre-evaporator system and flame stabilizers. The tests were stopped after a time, because there was no point in developing a system satisfying the claims of adjustment, for test conditions which were necessarily variable.

To study the possibilities of operating a ram-jet engine not only with liquid fuel, but also with *solid fuel*, a blower producing an air-speed in the range 15 to 35 m/sec was set up in front of a special combustion chamber. Streamlined steel grate boxes inside the chamber were loaded with about 40 kg of lump coal or briquettes, or with 40 kg of foam coal units, and ignited. These grate tests, in which the fuel was ignited by a primer composed of paraffin, lumber chips and thermite, showed that the grate bars either needed water cooling against the high temperatures or that they had to be made of highly fire-proof ceramic material to prevent melting. They also showed that the weight of the grate boxes was more than 30% of the total weight of the burner. Fig. 13 gives a picture of a water-cooled grate weighing 12.7 kg and holding 46.8 kg of lump coal. Foam coal units were mounted with devices of very small dead weight and



Fig. 12. Gas oil combustion test with pre-evaporator system in combustion chamber behind blower

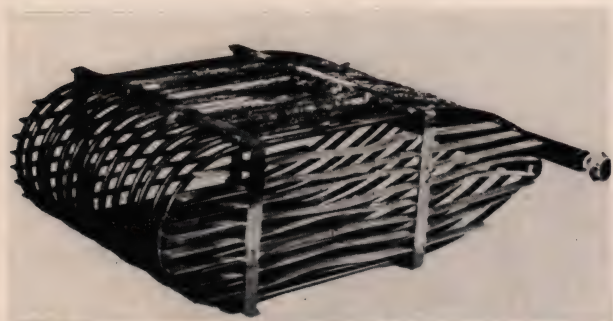


Fig. 13. A water cooled grate for lump coal combustion

burnt off in the free air flow. Their combustion rate, however, was much slower than that of lump coal. Only fractions of the combustion rate of 50 million kcal/m<sup>3</sup>h necessary for high thrusts could be realized. It also seems that the inertia of coal combustion will lead to tactical difficulties in aeronautical applications, for instance when switching from cruising to full powered flight.

The investigation of the disputed *altitude effects* on combustion was made by combustion tests with small hydrocarbon flames in the low pressure chamber of the DFS. They proved the combustion in higher altitudes to be more a problem of proper fuel supply to the flame zone than of reaction kinetics. The tests went down to a pressure corresponding to an altitude of 18,400 m, where the combustion of laminar-flow gases came to an end.

*Longitudinal sections of duct models* of one seventh of the normal size were tested in a smoke tunnel at flow speeds from 3 to 6 m/sec. The tests, which were run with varying nozzle areas and variable resistance grids in the combustion chamber inlet area, were primarily to show how far wind tunnel tests can

qualitatively imitate combustion flow conditions by inserting artificial resistances in cold engines.

Finally, duct model tests were run with variable nozzle areas and different inclination angles in a shallow water channel 26 mm deep at speeds from  $M = 0.3$  to 1.5 to study the development and the movements of the compression shocks in the diffuser. By sprinkling aluminium powder on the water surface, it was possible, moreover, to see any turbulence. Also, these tests had the same purely qualitative scope as the smoke tunnel investigations.

## 7. DEVELOPMENT OF DEFINITE RAM-JET PROJECTS IN CO-OPERATION WITH INDUSTRY

In spite of the really satisfying tests results, it was only in 1944 that an official work order for the development of the ram-jet engine was issued.

Negotiations for the co-operation of our experimental group with DORNIER had already been started as early as October 1942; conferences with the same aim were held in Spring 1943 between offices of the Air Ministry and the firm of JUNKERS. In May 1943 we got in touch with the firm of WALTER, which, in Hirschberg, had started in the meantime the design of a combined rocket ram-jet power plant. A little later, about November 1943, the experiences of our experimental group were handed over to FOCKE-WULF with the intention of winning their co-operation on the ram-jet domain.

All these efforts failed the first time either because *the firms* were blocked by orders and wanted to make pay off the money invested in the current classical developments, before accepting new engagements, or because *we* insisted on carrying on our own designs.

Among the possible backgrounds of these failures, there is one case which I ought to recall for its almost tragic-comical situation and the lesson it can give for the future. To be absolutely sure, the Ministry had in 1942 asked a prominent aerodynamical scientist for an expert's report on the new propulsion system. This overburdened man obviously overlooked a mistake he made by dividing the measured gross thrusts by the double ram pressure,  $\rho v^2$ , instead of by  $\rho v^2/2$ , so that the net free thrust, after deducting all the theoretical resistance coefficients, turned out to be too small. When we accidentally got hold of this report a long time later and had the opportunity to give (on 16th September 1944) a written correction, almost two years had been lost. Accidents like this probably were the reason, why, at a decisive congress on Jet Propulsion, in Berlin on 29th September 1943, the ram-jet tests were only mentioned during a discussion after the lectures. Even on 11th January 1944, at a meeting of the German Aeronautical Research Board, the flight tests and the hunter project with ramjets were designated as "not actual".

The reversal of this situation and the first positive co-operation with a firm began in December 1944, when we received a research order with the number OB 6429/262 from the "OBERBAYERISCHE FORSCHUNGSANSTALT, Oberammergau". The somewhat resigned sounding code name of this order was "HEIMAT-SCHÜTZER", i. e. home protector.

In those days of increasing fuel shortage and desperation, sudden rescue was expected from the ram-jet engine. The report of a meeting at the office of the

Chief of Technical Air Armament on 30th November 1944 which we received in January 1945, states literally: "At the introduction of the conference, the importance of the LORIN engine is to be pointed out . . . Because results have to be at hand as fast as possible, it will be necessary to start communication of experiences on a large scale. The development of subsonic propulsion devices shall be stressed, the supersonic speed range actually shall be encouraged only as far as the research stage is concerned."

In connection with this instruction, the possibility was considered of increasing the performance of the Me 262 by adding two ram-jet engines, which would be fed with the normal fuel of the TL-engine, the so-called J 2. Our design provided for two axially symmetric ducts of 1 m<sup>2</sup> area each, which were to be mounted above the wings and right over the TL-engines of the Me 262. Theoretical computation of this arrangement indicated that the following performance increases could be expected:

1. decrease of climbing time to an altitude of 10 km from 26 to 6 min without major changes of the starting attitude;
2. increase of the ceiling altitude by nearly 4000 m;
3. increase of maximum horizontal flight speed by 150 to 200 km/h.

The disadvantages, on the other hand, were a reduction of action range and flight time at a given fuel load. For example, at an operation altitude of 10 km the action range decreased from 1400 km to 470 km, and the time from 145 min to 50 min.

Work for this project had just been started, when the Supreme Command of the Luftwaffe issued a second work order in mid-December 1944 which gave us an administrative back-ground to the collaboration with the Aeronautical Research Establishment in Vienna. This collaboration had already existed, as a matter of fact, since April 1944. The LIPPISCH Delta-VI body, originally planned as a single-seat fighter, was intended to be built as a 3 L-device; this meant, according to the code of that time, as an "unmanned device, powered with solid fuel on the LORIN principle, the fuselage of which consisted mainly of explosives". The design provided for a central installation of the ram-jet engine with main diameter of 2000 mm. The possible thrusts computed for climbing at full powered flight were

$$T = 7300 \text{ kg at } 0 \text{ km altitude and } M = 0.81,$$

$$T = 2140 \text{ kg at } 12 \text{ km altitude and } M = 0.92.$$

At throttled level flight they were

$$T = 2300 \text{ kg at } 0 \text{ km altitude, } M = 0.81 \text{ and } m = 0.17,$$

$$T = 830 \text{ kg at } 12 \text{ km altitude, } M = 0.92 \text{ and } m = 0.23.$$

If an engineer in charge of the execution of the design work arising from the two work orders mentioned above could have felt perhaps still as uneasy as a tailor altering an outmoded garment, he was surely compensated by the third work order coming in from the firm of SKODA at the beginning of 1945. This concerned the design of a subsonic test plane with pure ram-jet propulsion, i. e. a really first project. Due to the general fuel situation it was thought to use, provisionally, foam coal, but the preliminary design already provided for alteration possibilities. The following construction details of this new craft, the

so-called "SKODA fighter Sk P 14 with SANGER-LORIN ram-jet", may be quoted here from the construction manual.

Dimensions:

Wing area	12.5 m <sup>2</sup>
Wing span	7.9 m
Length	9.5 m
Total height	4.2 m
Aspect ratio	5.

The centrally located power plant was to have a diameter of 1.5 m; the pilot's seat, as well as fuel tanks and tail construction, was placed above the landing skid beneath. The weight balance was:

Dead weight	1225 kg
Additional equipment	255 kg
Gross weight	1480 kg
Fuel	1200 kg
Crew	100 kg
Ammunition and oxygen	70 kg
Total weight	2850 kg.

The wing loading was therefore 228 kg/m<sup>2</sup>. The optimum operation time at an altitude of 13 km was to be 45 min at 640 km/h speed, the maximum speed near the ground about 1000 km/h, the service ceiling 18.5 km and the time to reach it (including take-off) 12.7 min. The climbing speed at ground level would have been 135 m/sec at 800 km/h flight speed. The plane was to take off with booster rockets and launching carriage after 200 m taxiing distance, the landing speed would have been 150 km/h at a landing weight of 1500 kg and needed 150 m of a gliding path.

In April 1945, in the very last days of the war, we finally got a commission from the firm of HEINKEL in Jenbach, concerning the design of a fighter of 3 tons flying weight with gasoline-operated central ram-jet engine and booster rockets for the take-off.

It was too late. At the end of the war all these developments were no further than the project stage or had just undergone preliminary tests. Civil aviation in peace-time, however, is less interested in high thrusts than in long action ranges and long flying times. This means that the ram-jet engine will be of interest for large aircraft as a main power plant only beyond the speed limits of  $M = 2.5$  reached nowadays by turbo-jet engines. Therefore, we do not want to close this paper without casting a glance at a project which was designed in the last days of the war and which deals with a ram-jet powered long-range aircraft for supersonic speeds, as it seemed realizable, conforming to the state of technical development in those days. Fig. 14 shows a sketch of the body, the estimated flight polars and the weight balance.

Although in those days further development was interrupted by the political situation, and although in the following years we could influence the coming achievements only indirectly in a consulting way, we are happy and satisfied by the news we learned some days ago, which shows what a high position the ram-jet principle has meanwhile attained in aeronautical engineering. Two

little notes in 1913 and 1941 had provided the initiation for these achievements. The reward may be seen by us in another recent little quotation stating that an unmanned ram-jet powered craft, "SNARK", had covered a distance of more than 3000 km.

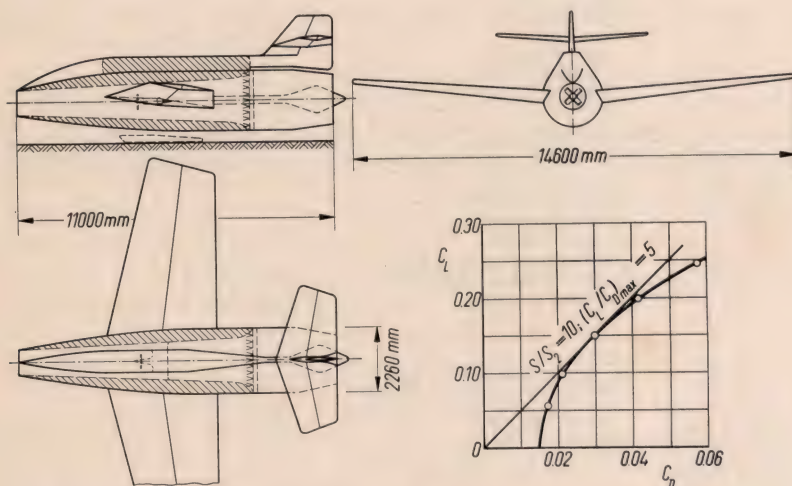


Fig. 14. Design of a ram-jet powered supersonic aircraft

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# HIGH-VELOCITY FREE-FLYING RAM-JET UNITS (TR-MISSILES)

Research Work at the German Army Ordnance Office 1936—1945

WOLF TROMMSDORFF \*

## INTRODUCTION

The conception of "missiles which are given an initial velocity in a gun barrel and which, after leaving said barrel, will be further accelerated by an additional rocket propelling unit using solid fuel", in connection with another conception, viz., "to reduce the percentage of oxygen carriers in the solid rocket propelling fuel and to supply instead the rocket combustion chamber with suitably compressed atmospheric oxygen" led in 1936 to post-accelerated artillery missiles as a special development of ram-jet propulsion.

The development of these missiles as high-velocity ram-jet propelled units was rendered possible step by step:

- a) By theoretical investigation of the working of ram-jet units, and by research into the influence of the individual flow cross-sections within the missile on thrust, specific net thrust and efficiency, at varying heating conditions in the combustion chamber and at varying flight MACH numbers;
- b) by the introduction of the multiple-shock supersonic diffuser suggested by the KAISER-WILHELM-INSTITUT FÜR STRÖMUNGSFORSCHUNG at Göttingen;
- c) by the development of diffusers fitted with intake cross-sections varying according to the MACH number;
- d) by the introduction of self-igniting propellant agents;
- e) by particularly intensifying the generation of mixtures;
- f) by overcoming one after the other the technical difficulties originating from the firing process with its high gas pressures and high accelerations;
- g) by the technical development of suitable fuel supply systems, and by working out a suitable fuel flow regulation.

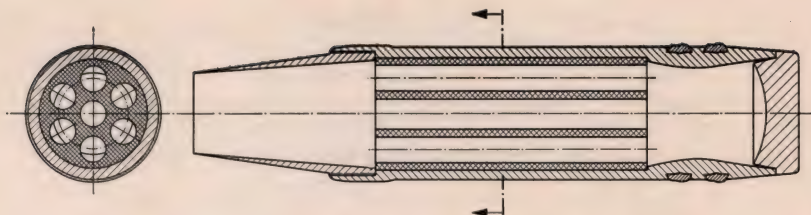
The results of these developments were checked either by observation of free-flying propelled missiles on an artillery range, or by testing small burning models of propelling units in pressurized supersonic wind tunnels.

At  $M = 3.0$  thrust efficiencies of more than 37% were recorded.

## 1.

In October 1936, the following proposal was submitted to the Army Ordnance Office:

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Fig. 1. 8.8 cm Tr-missile  $E_1$ 

"To fire a rocket-driven missile from a gun barrel and to ram the air at the head of the fast-moving missile for introduction into the rocket combustion chamber, either as a supporting mass, or as an oxygen carrying agent."

Following this proposal, different modifications of such a missile were constructed towards the end of 1938 and fired from an 8.8 cm anti-aircraft gun.

The test missile, fired at  $M = 2.5$ , carried in its cylindrical centre-part a compressed propellant powder charge with several longitudinal holes drilled through. This powder was so mixed that it was rather deficient of oxygen carrying agents and burned slowly, generating combustible gases.

During acceleration within the gun barrel, the missile was sealed at its rear end by a propulsion disk which could be dropped easily.

Tests were unsatisfactory. In some missiles the compressed powder charge was destroyed by the stress produced on firing. The remaining combustible powder gases, therefore, did not properly mix with the air streaming through the missile. Complete combustion was not obtained. Survey of the trajectory by means of a cine-theodolite did not prove beyond doubt that an effective thrust was being obtained. Today it is known that the use of solid fuel, technically even of coal, as a propelling agent is not fundamentally wrong or impossible. Regardless of the negative results achieved, these preliminary tests stimulated a new development, i.e. that of high-velocity ram-jet units.

## 2.

Fig. 2 shows the principle of action of a ram-jet.

The working agent (air) enters the ram-jet, designed either as channel-type (Fig. 2a) or, with identical axial cross-section contour, as ring-type (Fig. 2b), at 0 with the velocity  $w_0$  (flight MACH number =  $M_0$ ). In the diffuser between section 0, the entering section, section 1, the critical diffuser section (with passing the velocity of sound), and section 2, the end section of the diffuser, the air is now retarded as far as possible, whilst pressure and temperature increase. At 2, the addition of heat by combustion begins and this is terminated at 3, whilst in the nozzle from section 3, the cut-off of combustion, to section 4, the critical nozzle section, and section 5, the nozzle end section, the working agent is being expanded. A thrust is produced in the direction in which the missile moves.

Neglecting friction- and diffuser-losses and assuming that neither  $\kappa$ , the ratio of specific heats, nor  $R$ , the gas constant of the working agent, be affected by

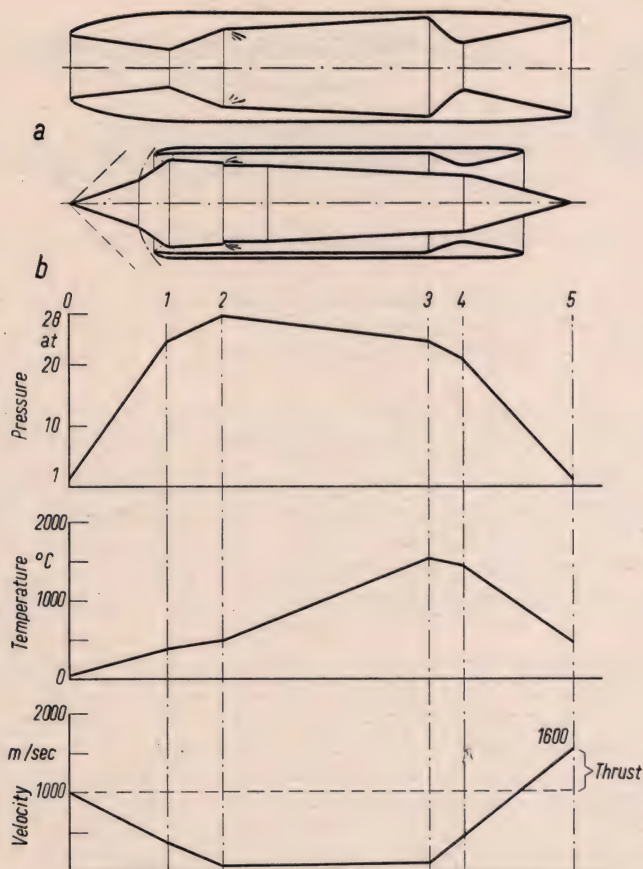


Fig. 2. Lay-out of ram-jet unit (characteristics of pressure, temperature and velocity within the unit)

- a) Channel-type (at low Mach numbers),      b) Ring-type (at high Mach numbers).  
 0 = Entering section      3 = Cut-off of combustion  
 1 = Critical diffuser section (throat)      4 = Critical nozzle section  
 2 = End section of the diffuser (fuel injection)      5 = End section

the addition of heat (combustion), some relations may be deduced describing the performance of a favourably laid-out propelling unit.

The maximum performance theoretically to be accomplished amounts to

$$(1) \quad \frac{T}{S P_0} = \kappa M_0^2 \frac{\varepsilon - 1}{\varepsilon},$$

$$(2) \quad C_T = \frac{2(\varepsilon - 1)}{\varepsilon} = \frac{T}{(\rho_0/2) w_0^2 S},$$

$$(3) \quad \eta = \frac{[(\kappa - 1)/2] M_0^2}{[(\kappa - 1)/2] M_0^2 + 1} \frac{1}{1 + (\varepsilon - 1)/2}$$

where

$T$  = thrust in kg,

$S$  = main frame cross-section area of the system in  $\text{m}^2$ ,

$P_0$  = surrounding pressure in  $\text{kg}/\text{m}^2$ ,

$\kappa = C_p/C_v$ , the ratio of specific heats,

$\varepsilon$  = the rate of heating up, defined by the quotient of the boundary velocities  $c_3$  and  $c_0$  of the gaseous working agent after and prior to the addition of heat,  $\varepsilon = c_3/c_0$ ,

$c_n$  = the boundary velocity at the point  $n$ , defined by the formula

$$c_n^2 = w_n^2 + \frac{\kappa g R \Theta_n}{(\kappa - 1)/2},$$

$C_T$  = thrust coefficient (analogous to the resistance coefficient and to the lift coefficient),

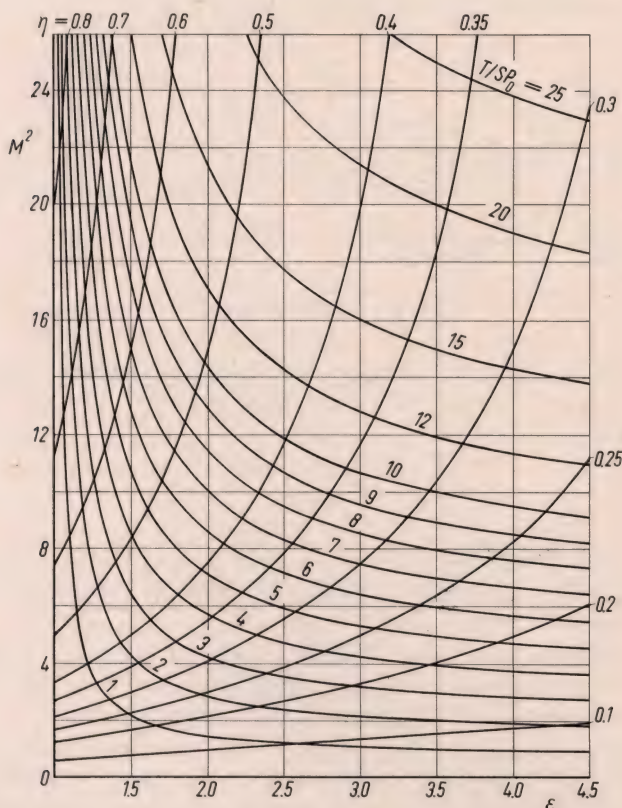


Fig. 3. Efficiency and specific net thrust in dependence upon flight Mach number and rate of heating

$\eta$  = efficiency of the system, defined by the quotient of the performance delivered as thrust to the moving rocket and of the overall working performance.

Fig. 3 shows the dependence of the efficiency  $\eta$  and the specific net thrust  $T/S P_0$  upon the flight MACH number and the rate of heating  $\varepsilon$ .

Fig. 3 reveals that in the higher MACH number range specific net thrusts and efficiencies may theoretically be achieved which should provide a wide field of use for the ram-jet propulsion method.

### 3.

Many difficulties had to be overcome in the tests and in the practical realization of the method. The high pressure recovery assumed theoretically, which is connected with the isentropic change of the kinetic energy of the entering working agent (air), was not realized.

If no special steps are taken (channel type), a normal pressure shock will occur before the intake aperture which, with increasing MACH number, greatly diminishes the efficiency of the pressure recovery (Fig. 4).

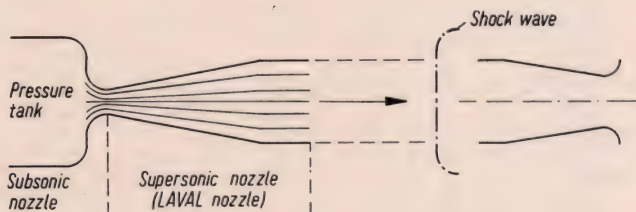


Fig. 4. Pressure recovery

But since the procedure only becomes promising at the higher MACH numbers, the whole work of development might have been in vain right from the beginning.

### 4.

The way out of this difficulty was provided in 1941 by the multiple-shock supersonic diffuser suggested by the KAISER-WILHELM-INSTITUT FÜR STRÖMUNGS-FORSCHUNG at Göttingen.

If the supersonic flow is led so that a series of oblique pressure shocks originate in it, then in such a system the pressure recovery in the supersonic part of the diffuser may be higher than in a normal pressure shock.

Fig. 5 shows the arrangement of shocks in a plane multiple-shock supersonic diffuser. The most favourable conditions of pressure recovery are obtained if the relative increase of pressure is identical in all shocks (OSWATITSCH law).

The theoretical efficiency of the pressure recovery in optimum two-dimensional multiple-shock supersonic diffusers is shown in Fig. 6.

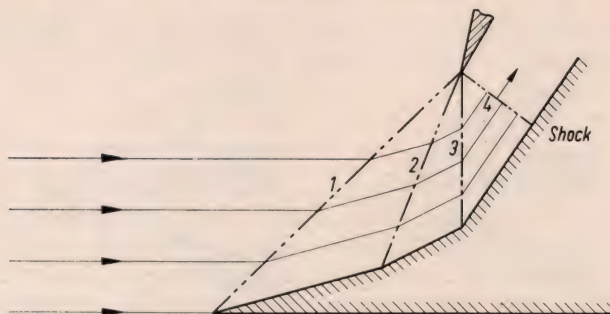


Fig. 5. Lay-out of a four-shock diffuser

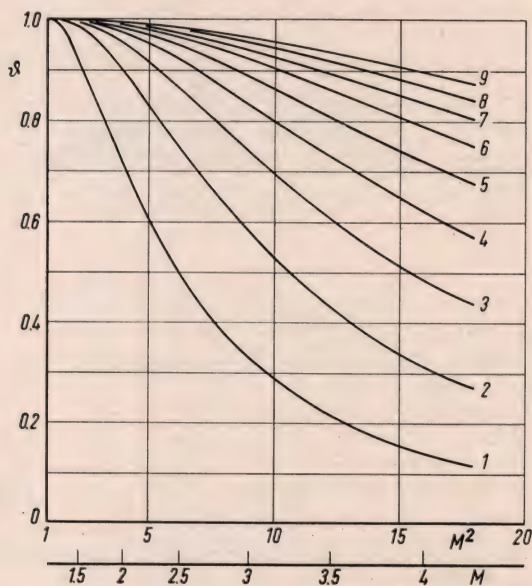


Fig. 6. Efficiency of multiple-shock diffusers in dependence upon number of shocks and Mach number

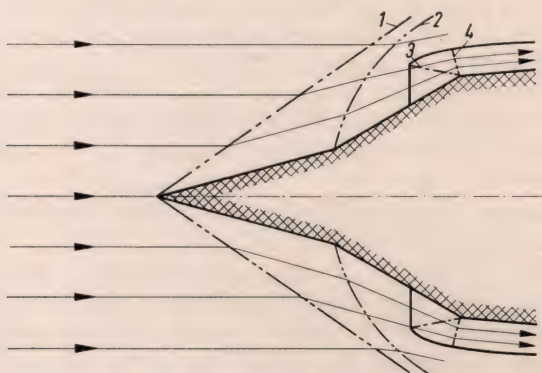


Fig. 7. Axially symmetric multiple-shock diffuser

The arrangements as used in practice are shown in Fig. 7. Here the oblique shocks were originated by a series of co-axial circular cones.

## 5.

The supersonic part of the multiple-shock diffuser is followed by the subsonic part, with the cross-sectional area increasing downstream. In this subsonic part, downstream of the throat, the normal shock terminates the system of pressure shocks. Depending upon the throttling of the critical cross-section at the end of the system, the final straight shock will advance towards the throat of the diffuser, resulting in a better pressure recovery.

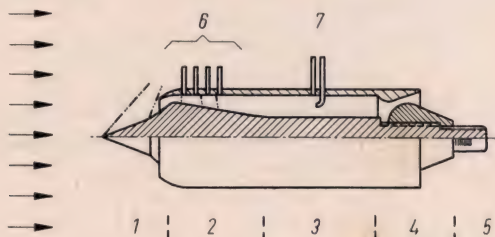


Fig. 8. Ram-jet model for tests in wind tunnels

- 1 = Multiple-shock diffuser
- 2 = Subsonic part of the diffuser with different positions of the straight shock
- 3 = Combustion chamber
- 4 = End nozzle with variable cross-section
- 5 = Measuring scale
- 6 = Measurements of pressure
- 7 = Measurements of mass flow and pressure

The classical test arrangement of OSWATITSCH for the investigation of these relations is shown in Fig. 8.

If, however, this shock approaches too near to the throat of the diffuser, the whole flow-diagram will become unstable. The shock moves upstream and places itself in front of the diffuser. In the combustion chamber the pressure will for some time be higher than the pressure behind the normal shock, which will now remain for a short while in front of the diffuser. Following this pressure drop, a reverse flow will occur for a short period, from the interior of the unit, until the pressure in the combustion chamber has decreased sufficiently. Then the back-flow will cease and the flow, together with the shock-system, will again be built up for a short while.

This phenomenon is repeated in a periodic fashion. Frequencies throughout the range of audibility were observed.

The appearance of this "diffuser humming" is detrimental to the performance of the propelling units. Thrust decreases and the resistance of the propelling unit increases. This "diffuser humming" was noticed when units were tested on the trial-stand as well as when tested in free flight. It can be avoided if the critical narrowest cross-section at the end of the combustion chamber  $S_3$  is

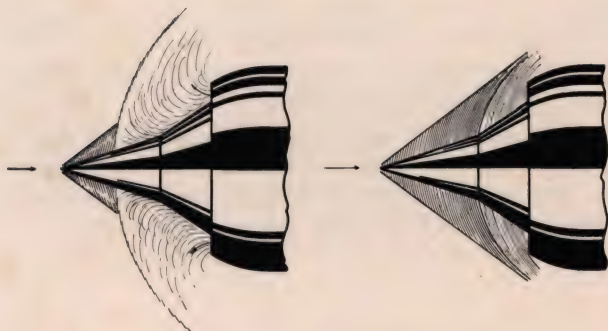


Fig. 9. Unstable flow at a multiple-shock diffuser

increased, but there is a reduction of the pressure recovery \*. "Diffuser humming" on one side and unsatisfactory pressure recovery on the other side are the "Scylla and Charybdis" when designing high-velocity ram-jet units.

Oblique blowing favours the appearance of "diffuser humming". For  $M > 3$  angles of pitch up to  $6^\circ$  are admissible. For  $M \approx 2$  angles of pitch must be less than  $3^\circ$ .

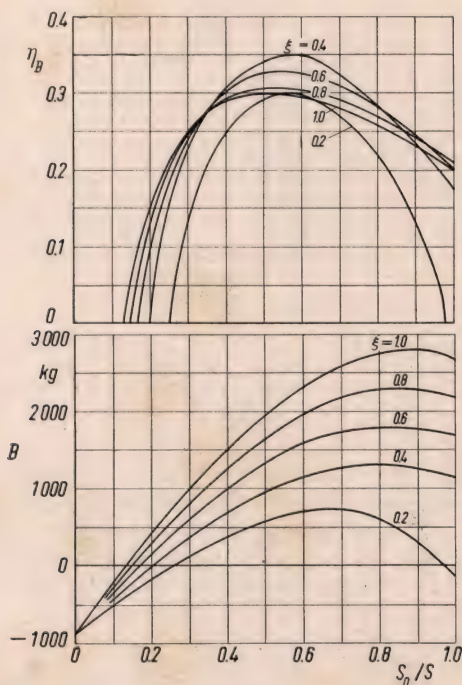


Fig. 10. Specific acceleration thrust  $B$  and efficiency  $\eta_B$  (of a tetraline-heated ram-jet unit) in dependence upon the fuel number  $\xi$  and the ratio of aperture  $S_0/S$  for  $M = 3$

\* The relations mentioned above between quality of diffuser and stability of flow were, for the first time, clarified theoretically and experimentally in 1941 by K. Oswatitsch.

An inadmissible increase of fuel delivery and a correspondingly increased generation of gas in the combustion chamber will also lead to "diffuser humming". This fact has to be considered in connection with the development of thrust regulation by governing the delivery of fuel.

When designing a propelling unit, the two parameters which may be deliberately chosen are the aperture ration  $S_0/S$  and the rate of heating of the working agent. The dependence of the performance of a system upon the choice of these two parameters is shown in Figs. 10 and 11.

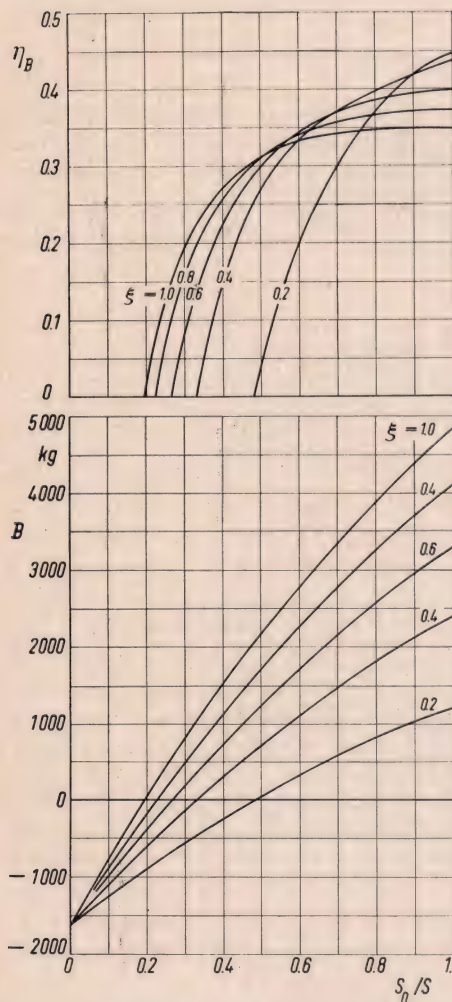


Fig. 11. As at Fig. 10, but for  $M = 4$

It emerges from Figs. 10 and 11, that a propelling unit may be designed for either maximum thrust or maximum efficiency. Generally, the maximum efficiency is obtained when a considerable surplus of air is available.

As visible proof of the thrust, missiles were required to accelerate from the muzzle velocity  $v_0$  to the maximum velocity reached on burning out.

The determination of the most favourable  $S_0/S$  and of most suitable rate of heating, shown in Figs. 10 and 11 for certain MACH numbers, had to be performed for the total MACH number range under consideration. It was found that the  $S_0$  for maximum efficiency increases with increasing MACH number. Thus, for a propelling unit, designed to work within a certain MACH number range with maximum or almost maximum efficiency, it is necessary for  $S_0$  and the mixture ratio  $\xi$  to be adapted to the relevant working MACH number.

## 6.

How a diffuser can be built with  $S_0$  varying according to the MACH number without incorporating moving elements is shown in Fig. 12, illustrating a simple two-shock diffuser which is blown at varying MACH numbers. It emerges from the flow-line diagram, that the intake cross-sections  $S_0$  vary with varying MACH numbers. Furthermore, it appears evident that by a suitable choice of point  $E$  (diffuser edge), a variable  $S_0$ -MACH number ratio may be fixed.

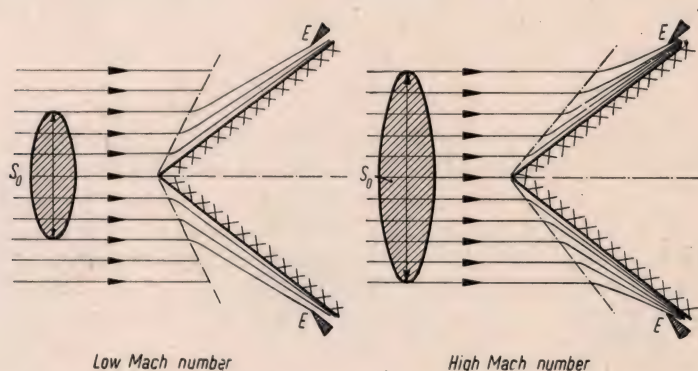


Fig. 12. Axial supersonic flow at a three-dimensional two-shock diffuser at different Mach numbers

As far as the diffusers most employed in practice with several shock-cones are concerned (Fig. 7), the analysis of the flow diagrams at varying MACH numbers takes more time, but it supplies a larger amount of possible variations in geometry and allows the dependence of  $S_0$  on  $M$  to be adapted even more exactly to the required optimum working conditions.

## 7.

The special construction of these propelling units as artillery missiles produced a number of particular problems. Since, in the case of artillery missiles stabilized by rotation, the total length is limited — at the utmost — from 4.2 to 4.9 calibres (longer projectiles cannot be held stable in flight and in

track by rotation alone), the volume of the missile is restricted when the calibre is given.

This restriction of the total length of the missile to a few times the calibre results in a minimum calibre for proper long distance missiles. A pre-condition is also set by the requirement that for long distance missiles the maximum velocity has to be reached in about 25 km flight at altitude. Since ram-jet propelled missiles are specifically light bodies (a conventional 28 cm missile weighs about 350 kg, a ram-jet propelled missile of equal calibre about 170 kg),

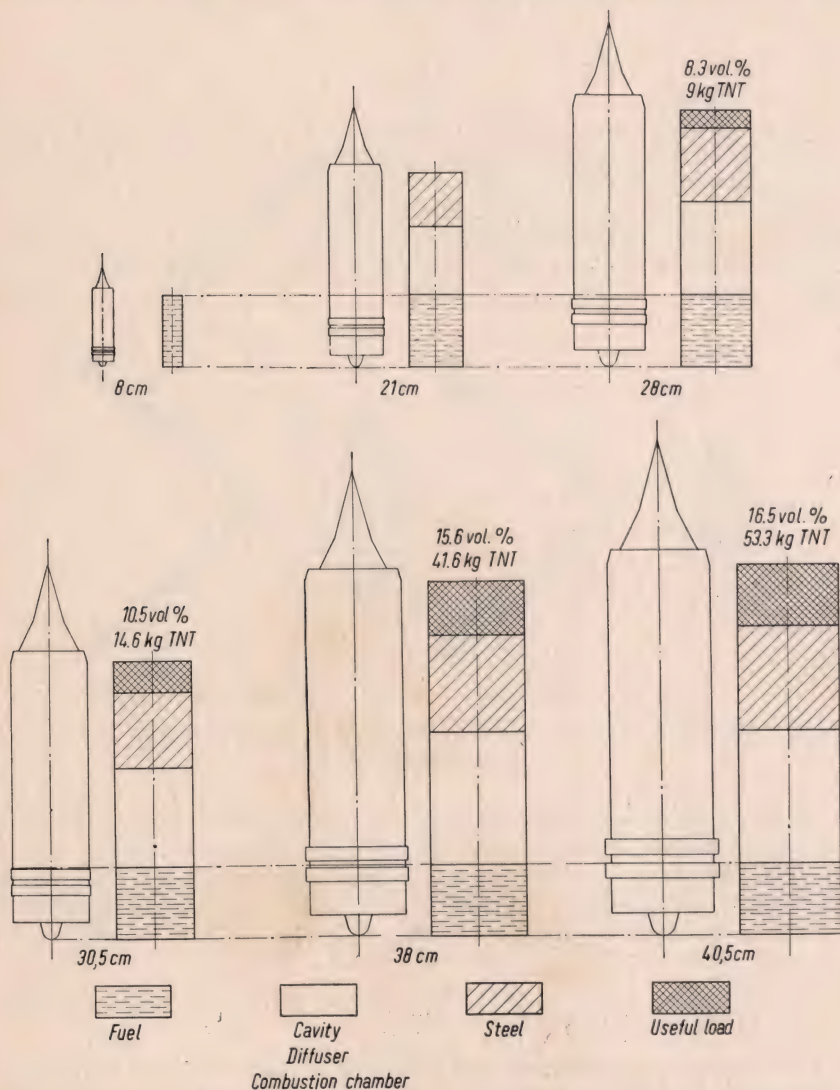


Fig. 13. Distribution of volume within the missile; possible useful loads in long distance missiles stabilized by rotation

the acceleration desired from the additional ram-jet propulsion would, compared with the conventional missile, not result in an appreciable increase of distance covered, since the combustion would terminate prematurely in dense atmosphere near the earth. In order to obtain a sufficiently late final point of combustion, about 26 g of fuel for each  $\text{cm}^2$  of the main-frame cross-sectional area are required. The fuel tank, filling the whole cross-section frame area, would have to be 28 to 30 cm high. A missile of 9 cm calibre can, therefore, under no circumstances be designed as a long distance missile. The fuel required merely to reach a favourable height for terminating combustion, would fill its total volume. There would be no room left for steel, useful load, diffuser and combustion chamber. Large series of missiles, designed to the last detail, supplied the necessary data for the proper distribution of the volume available.

Higher flight MACH numbers reduce the volume requirements for gas-filled spaces within the missile, for the diffuser, combustion chamber and exit nozzle. Long launching tubes reduce volume and weight of the steel required.

A summary of the technical possibilities is supplied by Fig. 13. The distribution of volume in a ram-jet propelled long distance missile of varying calibre, stabilized by rotation, is reproduced schematically. It is evident that the carrying of a useful load only becomes feasible for calibres greater than 21 cm. But if the calibre is 28 cm about 40% of the useful load carried by the conventional missile of equal calibre can be conveyed.

## 8.

Development work, therefore, had to strive in the first instance for reducing the space of the combustion chamber, or for increasing the specific combustion chamber efficiency.

When a definite combustion chamber performance is required, the volume of the combustion chamber depends upon the time which elapses between the moment when the two combustible components meet, and the more or less complete termination of the combustion process. Schematically, this time may be divided into the period of forming the mixture, of ignition, and of combustion.

A thorough mixing of the components can be obtained either

- a) by the two components being in a gaseous state and flowing together through a tube. The turbulent flow within the tube will lead to the mixture being formed, or
- b) the one component being in a gaseous state, the other component of much higher specific weight being liquid. The liquid is shot out of very numerous and very small injection nozzles into the combustion chamber in such a way that, within a very short time, two components are locally correctly mixed.

In many known propulsion units ignition is effected by a hot ignition source (flame behind a flame-holder) heating up the ignitable mixture by exchange of matter, by heat conduction or by radiation. This phenomenon of turbulent mixture ignition took longer and required rather large spaces. Within the MACH number range of high-velocity ram-jet units, air temperatures at the end of the diffuser are so high that spontaneous self-ignition of injected little drops appears likely. In order to find suitable fuels which, in the range from 500 to

800° K would have self-igniting times of between 1 and 3 msec, the self-igniting temperatures and times of many liquid fuels were obtained by means of a simple test apparatus. The size of the drops and their velocity will also affect the self-ignition temperature so that very small and very fast drops, flying individually, will ignite less efficiently even though the air constants be unchanged. An ignition mechanism was developed (Fig. 14).

The drops enter with medium velocity (50 to 80 m/sec) into an ignition pit separated from the general air stream. In this pit, the fuel concentration is much above the stoichiometric concentration. The drops leave the ignition pit burning and join the fresh-air stream which is moving at higher velocity (60 to 120 m/sec), so that the velocities of drops and of the air are added vectorially. The trajectories of the drops are so governed that, as shown in Fig. 14, they pass right over the desired part of the combustion chamber cross-section. Suitable fuels had to be chosen in order to obtain low combustion times. Conventional fuels, such as mineral oil destillates, were not always the most suitable ones. Low molecular weight, unsaturated and unstable compounds show for reasons which are not always clear and obvious, a faster reaction up to a final high percentage combustion.

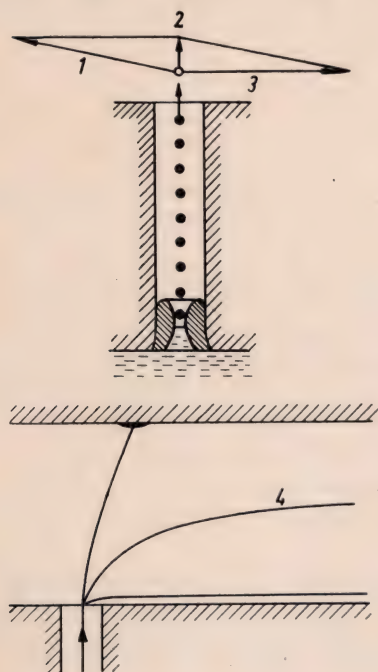


Fig. 14. Lay-out of fuel injection

- 1 = Relative velocity between fuel drops and air
- 2 = Velocity of the fuel drops
- 3 = Air velocity
- 4 = Satisfying trajectory of the fuel drops

The following fuels were systematically tested:

Carbon disulphide,

Carbon disulphide mixed with medium DIESEL oils (boiling point 200 to 250° C),

Propane,

Carbon hydrides of normal benzines,

Di-ethyl ether,  
Thioether,  
Zinc di-ethyl,  
Butadien,  
Acetone,  
Tetraline,  
Dekaline,

several technical DIESEL oils, particularly generated by lignite coking,  
as ignition and additive components:

Dichloracetylene,  
Amylnitrate,  
Amylnitrite,

the latter two mixed with carbon hydrids.

These investigations were neither fully completed nor have the test results been completely evaluated as far as physics and chemistry were involved. All these endeavours resulted in achieving, experimentally, specific net performances in the combustion chamber of  $1.5 \text{ kcal/cm}^3 \text{ sec}$  or 3000 HP and more mechanical performance for each litre of combustion chamber volume. These figures already approach the specific net performances of well-designed liquid-fuel rockets.

## 9.

The firing from a gun barrel puts particular demands on the ruggedness and construction of the units, demands which must accord with the aerodynamical requirements. Very many modifications of the form of the connecting elements between the central body and the outer shell were tested.

Cast, milled or welded connecting links without stress transmission or with only a slight load centered the front part of the central body and the front part of the shell against each other (Fig. 15).

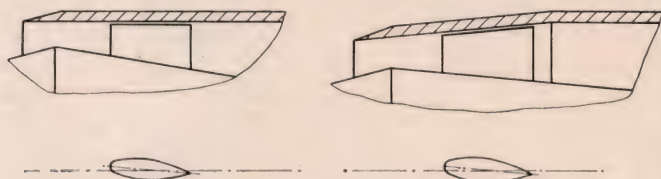


Fig. 15. Connecting links at the head, milled from full material

The link at the rear which transmitted the stress, was built as a drilled plate ( $E_3, E_4$ ), as massive milled diagonal links ( $C_3$ ), or in lantern-shape ( $E_5$ ) (Fig. 16).

The stresses to be transmitted were kept small due to the fact that the relation of the weight of the cover shell to the annular cross-sectional area of the shell under pressure was like the relation of the weight of the central body to its cross-sectional area under pressure. A large number of designs was tested, and each time the designs had to be rather close to the breaking points of the material. The decision as to whether the connecting links were sufficiently strong could only as a rule be established from results of the firing tests.

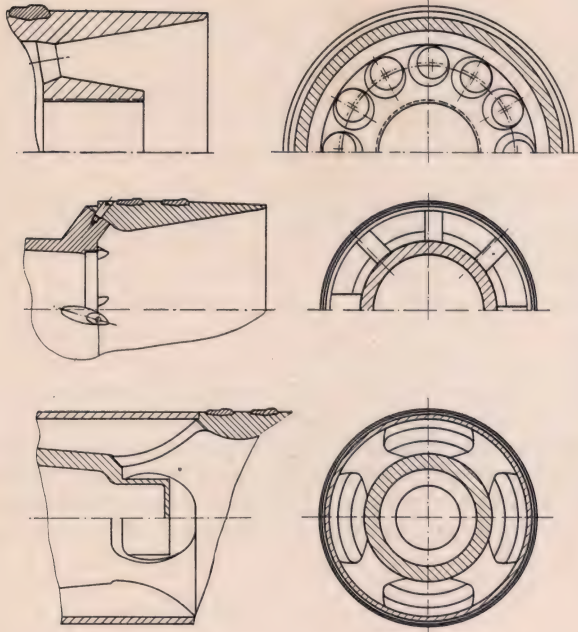


Fig. 16. Connecting link at the rear for transmitting stresses

10.

Whilst within the gun barrel, the propelling unit had to be sealed against the passage of high-pressured powder-gases by an easily detachable propulsion disk. It was required, and achieved, that the propulsion disk was to fall off at the

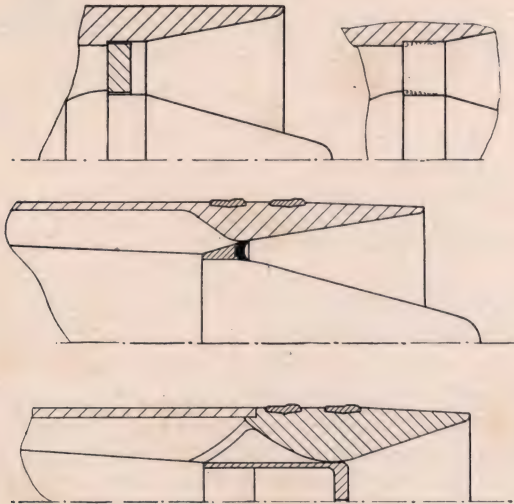


Fig. 17. Designs of propulsion disks

latest 15 or 20 m after leaving the gun barrel. The propulsion disk had to transmit considerable stresses. It had, therefore, to be supported by a sufficiently large area and "propulsion disk step" disturbed the flow just near the LAVAL nozzle. Very many different propulsion disk designs were tested (Fig. 17).

In missiles  $E_3$  and  $E_4$  the propulsion disk, which fell off easily and reliably, was made of the highest quality chrome-nickel steel and attached with paraffin. It was, however, very detrimental to the flow in the throat of the LAVAL nozzle. In later designs, the propulsion disk step was moved to the convergent part of the flow ahead of the throat of the LAVAL nozzle. This reduced the detrimental effect on the flow, but the disk was larger and longer and was more difficult to throw off ( $C_3$ ). The rather successful missile  $E_5$  had a very large propulsion disk, with a favourable located step which could be jettisoned.

## 11.

With rotation-stabilized missiles, care had to be taken, by fitting suitable blades in the fuel tank that, at least on leaving the gun barrel, the liquid fuel took part fully in the rotation of the missile, since otherwise the missile, filled with non-rotating fluid, would have become very unstable.

Fuel delivery with the necessary pressure at the injection nozzles was obtained from the centrifugal force of the rotating fuel together with pressure gas tanks or with powder charges generating pressure gas. Regulation of fuel delivery was so arranged ( $C_3$  and others), that the propulsion unit was set for a constant axial acceleration. A sliding weight held by springs governed the fuel delivery (Fig. 18). Since the spring-held sliding weight is never centered

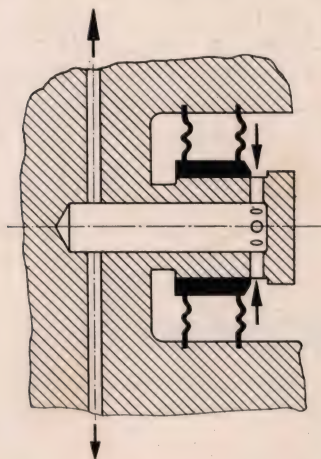


Fig. 18. Control of fuel supply by a sliding weight held by springs

exactly in the axis of rotation of the missile, very strong centrifugal forces which hinder the free movement to and fro may occur, if care is not taken that the central cylinder, on which the weight slides, has nearly the same specific weight as the fuel (material: plastics) and at its end is elastically attached so that it can move laterally. The technical manufacture of very many very small fuel injection nozzles (Fig. 19) was rendered economical by a pressing process.

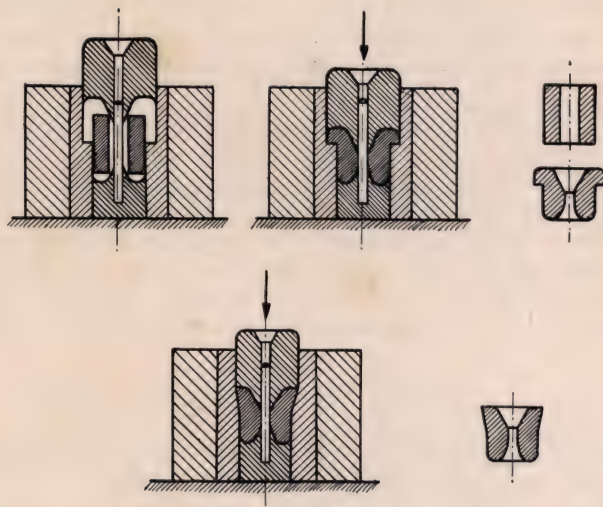


Fig. 19. Machining of injection nozzle

## 12.

The following compilation gives a review of the propelling units designed and actually constructed:

Propulsion units of the *A*-series were to receive their initial velocity from a detachable solid-powder rocket. Propulsion set *A*<sub>4</sub> was carefully designed (Fig. 20) but not built. Propulsion sets of the *A*-series, when used in action, will presumably have to be equipped with some sort of tele-guiding device, since the mechanism of the powder-rocket, as well as that of dropping, will lead to inadmissibly large dispersions.

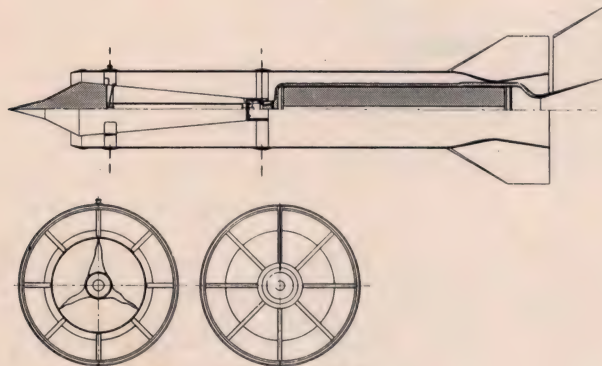


Fig. 20. Ram-jet unit *A*<sub>4</sub>

## 13.

The propulsion sets of series *B* were, by means of a very long, smoothly accelerating launching barrel with medium powder-gas pressure, designed to counteract the reactions which were evidently experienced upon the firing devices by the introduction of ram-jet units.

The designs of the *B*-series were those which complied best with the special concept of firing high-velocity ram-jet units out of gun barrels.

It was found that the firing of high-velocity ram-jet units out of conventional gun barrels has to be considered as an improvised emergency solution. If the gun and ram-jet missile are regarded as a constructional unit, it is obvious that the technical design of the gun must be influenced by the nature of the missile. For long-distance missiles or guns with a high  $v_0$ , an attempt was made to obtain, by corresponding construction of the missile, a high mass load in relation to the calibre cross-section. The higher the specific mass load, the longer the distances which may be covered with the conventional missile under otherwise equal conditions (supremacy of the "long missile" compared to the "ball missile"). High mass load on the cross-section, however, has disagreeable effects upon the construction of the gun. The gas pressure within the barrel and the thermal and mechanical stress upon the gun increase in proportion to the cross-section mass load.

In the case of a ram-jet missile, however, the construction of a high cross-sectional mass load is unnecessary and even detrimental.

The ram-jet driven missile covers the distance not by the kinetic energy of its inert mass, but by the continuous supply of energy from the ram-jet combustion. Under certain circumstances, a ram-jet long-distance missile may require but a very small cross-sectional mass load (provided the terminal point of combustion is chosen sufficiently high — about 20 to 25 km — on the ascending branch of the flight path). But the conventional long-distance missile requires high powder-gas pressure and such high pressure would render the construction of the ram-jet unit more difficult. In case of high gas pressure, the small volume available is particularly restricted by the accumulation of steel near the thrust nozzle outlet.

With high gas pressures, the difficulties of obtaining reliable release of the propulsion disk will increase, and the difficulties associated with the strength of the connecting links between central body and outer shell become too great.

Low cross-sectional mass load as well as low gas pressure were therefore aimed at in the design of the missiles of the *B*-series. Since with these devices the maximum recoil load on the ground can be kept within reasonable limits, the expensive and clumsy installation of recoil-absorbers and -springs may be dispensed with. Furthermore, since shooting is preferably in the upper angle-section, the recoil forces can be taken up, as in the case of trench mortars, by an anchored bottom-plate. Thus new super-long trench mortars without any provision for damping or absorbing the recoil, which is transferred to the soil by a bottom-plate directly and without damping, may be used for launching. Slow firing-off accelerations and slow initial velocities in the range of  $M \approx 2.5$  require very long launching tubes. Gas pressures in the tube will be rather low: 150 to 500 atm. The launching devices for the *B*-series (Fig. 21) were therefore called "super-long mortars".

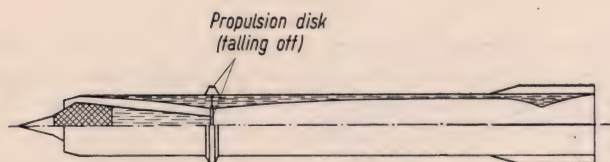


Fig. 21. Ram-jet unit  $B_2$

Owing to the relatively great length of the launching tubes, in transport they have to be divided into several sections which can be easily put together again in the firing position. The ability to adjust the assembled tube-sections is indispensable.

The missile itself appears but little different from those of the *A*-series. A propulsion disk in the vicinity of the missile's centre of gravity facilitates the static design of the missile.

#### 14.

Fig. 22 shows the ram-jet unit  $C_3$  of the *C*-series, which was built for firing from the German 28 cm gun K 5.

The careful exploration of several designs of the *C*-series revealed that such missiles of less than 21 cm calibre could not carry any additional useful load.

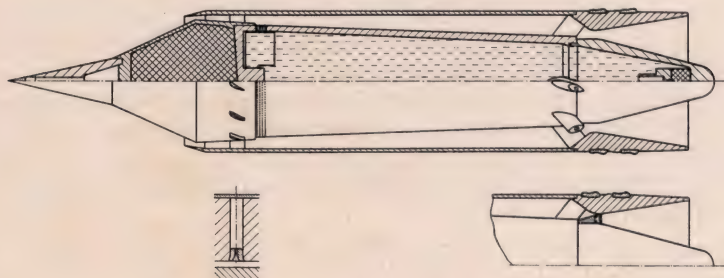


Fig. 22. Ram-jet unit  $C_3$

The initial velocity amounted to 1223 m/sec. Fuel delivery was effected by the burning of a powder charge, placing the fuel tank under a pressure of 180 atm. After passing through an acceleration-regulating device, the fuel was ejected through 480 nozzles. The combustion chamber was about 60 cm long. The temperature of the air, when meeting the fuel, was about 700° C. With this unit the choice of a suitable propelling agent was less restricted; carbon disulphide was rejected and medium DIESEL-oils and tetraline were used as fuel. The propulsion unit developed a thrust of about 2000 kg, having a total weight of 170 kg. The efficiency was 47%, the air resistance of the unit already having been deduced from thrust. Such a system may, with its fuel reserve, increase its velocity to 1860 m/sec and then cover distances of up to 350 km in free ballistic flight.

The author was not in a position to carry out tests with this unit. It has become known, however, that during tests with this unit at other places the performances mentioned above were in fact obtained. Only the cooling of the connecting links gave trouble.

15.

The *D*-series dealt with studies on inter-continental guided missiles. The missile  $D_{6000}$  which was designed for a distance of 6000 km, was carefully investigated (see Fig. in Zeitschrift für Flugwissenschaften 2 (1954), p. 239).

In this case the method by which the missile was to be brought to the launching altitude and launching velocity required was left open. A launching altitude of 14 km and velocities at this altitude of 200 m/sec had been assumed as a precondition. By means of two detachable launching rockets (boosters) at the wing-tips, the dropping missile was to be accelerated near the ground to  $M = 2.8$ . At  $M = 2.8$  near the ground the ram-jet unit was able to effect the further acceleration to the working MACH number of 4. This MACH number was reached at an altitude of 24 km; the missile would then proceed about 5000 km under its own power and would also be able to glide another 300 km. Practical tests were not undertaken with the missile  $D_{6000}$  nor with models, since by the end of 1944 no German aircraft was able to carry the missile in hump-back flight up to the required launching altitude and velocity.

16.

The *E*-series dealt with experimental missiles which were not intended for practical use, but for demonstrating the behaviour of free flying units and for supplying research-data. About 260 missiles of the *E*-series (Fig. 23) were built and fired-off.

The fuel, in this instance carbon disulphide, was stored in the centre of the unit. Since the unit on leaving the gun was given a certain rotation, the fuel was rotated by suitably fixed blades which transmitted the rotation of the

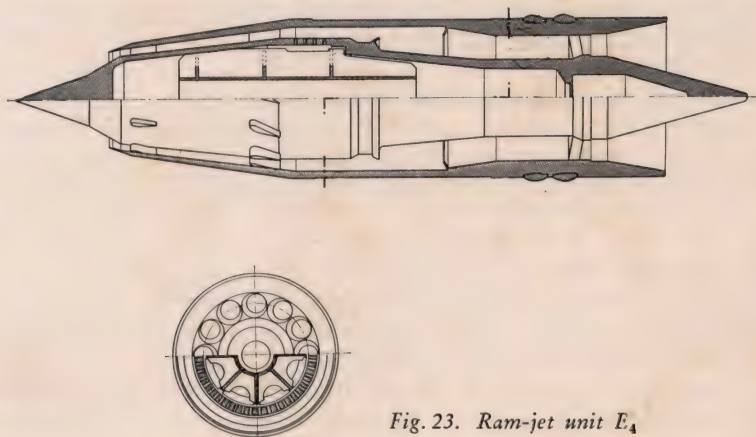


Fig. 23. Ram-jet unit  $E_4$

body to the fuel. The centrifugal force of the fuel, together with the gas pressure from a small gas-container, forced the fuel through the injection nozzles. Prior to the test, all injection nozzles were sealed gas-, air- and moisture-tight by a thin film of collodium, which was destroyed on discharge, thus opening the fuel nozzles. The injected fuel, distributed by 160 injection nozzles, ignited itself. The combustion chamber of only 25 cm length was fully adequate for a 98% chemical reaction. The gases passed through exit nozzles to the open. Such a propulsion unit showed the following performance: Specific net thrust about 2.3 kg/cm<sup>2</sup> cross-sectional nozzle area, i.e. the set developed a thrust of about 400 kg. The total weight was 28 kg. Rate of efficiency reached was about 37%. Initial velocities were 960 to 1020 m/sec; final velocities of 1460 m/sec were obtained after a burning period of 3.2 sec.

The ram-jet unit left the gun barrel of a 15 cm gun with an angle of 6°, so that the whole trajectory could be followed by ballistic survey-methods (cinetheodolite, acoustical survey and hit-observation). From the results of the survey, conclusions were drawn about the working and the properties of the units. Dummy missiles, filled with water instead of liquid fuel, made examination of the resistance properties of the missile possible.

## 17.

This research, at that time off the beaten track, hardly noticed and carried on with very little means, would have been dropped completely, if the person involved had not received scientific assistance and encouragement. He regards it as an honourable duty to express his thanks for this.

The ideas, then very new, were in the initial stage of their development from 1935 to 1938 sponsored by Prof. F. DRESCHER-KADEN, Göttingen, Director Dr. M. ESTERER of SIEMENS Ltd., Berlin, President Dr. SEIDEL, MATERIALPRÜFUNGSAMT, Berlin, who rendered most generous financial assistance, and Vice-Admiral (retd.) v. REUTER. The master mechanics GRÜNERT and RAPPE were untiring in making models without pay. The first practical results of this preliminary work were examined by Major Dr. SCHMAGER, Army Ordnance Office (WA PRÜF I) at the artillery range Kummersdorf. Looking far ahead, General BECKER and Prof. Dr. WINKHAUS, Technische Hochschule Berlin, ordered work to be continued at Kummersdorf artillery range. Dr. GLIMM (deceased) was for many years the faithful sponsor at Kummersdorf artillery range.

Later on, sponsorship concentrated upon scientific advice and assistance. Here it was the Göttingen circle, first of all Prof. Dr. PRANDTL, Prof. Dr. BETZ, Dr. LUDWIG and Dr. OSWATITSCH. The contribution rendered in particular by Dr. OSWATITSCH by his experimental and theoretical investigations of the properties of multiple-shock supersonic diffusers, is well known from his publications. At the same time, advice and help was proffered by the scientists around Prof. Dr. BUSEMANN, Braunschweig. Particularly valuable suggestions regarding questions of combustion and matters of physics and chemistry were supplied by Prof. Dr. ERNST SCHMIDT and Dr. DAMKÖHLER (deceased) at Braunschweig and by Prof. Dr. EUKEN (deceased) at Göttingen.

Working on inventions in a new field entails a heavy burden. The enthusiasm conveyed by the new ideas and by the new possibilities help the

inventor to overcome this burden, yet all disappointments and all set-backs have to be borne without complaint by the companion. I express my thanks in remembrance to Mrs. ROSEL TROMMSDORFF (deceased) who for 20 years shouldered this burden.

## DISCUSSION

Prof. Dr. QUICK (Aachen): Dr. TROMMSDORFF, you have explained to us how difficult to handle and how sensitive some of the shock diffusers were, and what solutions to make them insensitive to MACH number changes suggested themselves. What about the sensibility of multiple-shock diffusers to changes of the lead-angle?

Dr. TROMMSDORFF: Near the point of maximum pressure recovery the sensibility of multiple-shock diffusers to changes of the lead-angle will increase with decreasing MACH number. At  $M = 3.2$ , the flying missile showed deviations of the missile axis from the direction of flow amounting to about  $7^\circ$  as could be established by careful analysis. Between  $6^\circ$  and  $7^\circ$ , the missile could be easily heard to be becoming unstable due to a characteristic humming sound. At  $M = 2$ , lead-angles of  $3^\circ$  appear to constitute a deadly danger.

Dr. KURZWEG (Silver Spring, Maryland): To begin with, I want to express my pleasure at having been given the opportunity to listen for the first time to a comprehensive report delivered by Dr. TROMMSDORFF. Though both of us were working in the same office, the Army Ordnance Office, I have ignored up to now what Dr. TROMMSDORFF has in fact done. We had never been able to exchange our experience and our ideas. During the last two years of the war, we examined the problem of pressure recovery and obtained highly satisfactory results. As far as I can remember, we came to the conclusion that this principle looked promising for large missiles only, but would be of little value for small ones. Now I would like to ask a question which presumably would have been answered 15 years ago, if we had met each other. You said that the initial powder unit did not fly. Had you, at that time, calculated the missile performance correctly and did you have the proper coefficient of stability?

Dr. TROMMSDORFF: Regarding the research work, I may claim perhaps one merit, i.e. to have ploughed through all errors and mistakes possible in designing high-speed ram-jet propelling units. The first missile was an unfortunate and badly conceived design which today I would like to pass into oblivion and to cover with silence.

Dr. KURZWEG: Nowadays, the multiple-shock diffuser is generally known as the OSWATITSCH diffuser. It might be of interest to hear something about the history of this idea.

Dr. TROMMSDORFF: During our investigations in summer 1940 it became obvious that the overriding problem in the development of ram-jet propelling units for high MACH numbers would be to develop supersonic diffusers with high pressure recovery working in such a MACH number range. Simultaneously it was invariably found that, with the auxiliary means known at that time, nothing better than the unsatisfactory pressure recovery of the simple straight

pressure shock might be obtained. Following my suggestion, the Army Ordnance Office asked the INSTITUT FÜR AERODYNAMIK (Head: Prof. Dr. A. BUSEMANN, Assistant: Dr. G. GUDERLEY) of the LUFTFAHRT-FORSCHUNGS-ANSTALT at Braunschweig and the KAISER-WILHELM-INSTITUT FÜR STRÖMUNGS-FORSCHUNG at Göttingen (Head: Prof. Dr. L. PRANDTL, Assistant: Dozent Dr. K. OSWATITSCH) to examine whether there existed gas-dynamical lay-outs offering, for  $M > 2$ , a better pressure recovery than the one obtained by a straight pressure shock. The replies of the two research-teams entrusted with this task arrived almost at the same time.

The investigations carried out in Braunschweig resulted in arrangements of diffusers without central bodies, e. g. the BUSEMANN ring. With these arrangements a combination of oblique and straight shock-waves and a higher pressure recovery were obtained.

The first hint to increase pressure recovery by combining oblique and straight shock-waves, was given to me by Prof. PRANDTL. I am not aware to what extent this information was preceded by discussions within the Institute.

From 1941 onwards, the systematical investigation and development of diffusers with two and more shock-waves on conical central bodies was performed by K. OSWATITSCH. In particular, OSWATITSCH tested and developed:

1. the plane triple-shock diffuser with suction of the boundary layer prior to each shock ( $M \approx 3$ );
2. the exact deduction of a law regarding the lay-out of multiple-shock diffusers of maximum efficiency (by practical tests the same law was found independently and simultaneously in Kummersdorf);
3. axially symmetric triple- and quadruple-shock diffusers within the MACH number range 3
  - a) with suction of the boundary layer,
  - b) without suction of the boundary layer,
  - c) with small or vanishing wave resistance;
4. the phenomenon of diffuser humming was observed for the first time and interpreted and systematically investigated by OSWATITSCH.

Furthermore, in Göttingen, Prof. PRANDTL, Prof. BETZ (who in particular drew the attention for the first time to the possibilities of supersonic compression in compressors by means of multiple-shock diffusers) and Dr. LUDWIG took a leading part in the discussions and investigations on the problem of supersonic diffusers.

I do not know how far these gentlemen were individually responsible for first laying down this principle, since at that time questions of priority were considered as being of minor importance. Development progressed quickly, the assistance offered by the Braunschweig and Göttingen teams proved successful and efficient so that today, 15 years later, it would be extremely difficult to reconstruct the history of development of the very useful multiple-shock supersonic diffuser in full detail.

# ON THE HISTORY OF THE DEVELOPMENT OF THE SCHMIDTROHR

PAUL SCHMIDT \*

## 1. INTRODUCTION

In his excellent paper of 1946, "The Pulsating Jet Engine — Its Evolution and Future Prospects" L. B. EDELMAN provided a review on the development of the pulse-jet<sup>1</sup>. I would like to emphasize his remarks on the work of KARAVODINE and the ideas of MARCONNET, in which it is shown that the principle of intermittent combustion had been introduced as early as 1908. KARAVODINE's turbine constructed and operated in 1908, constituted a remarkable technical achievement, and mention must be made of BARBEZAT who, 50 years ago, succeeded in measuring the pressure distribution of an intermittent combustion. In my first publication in 1948 reference was made by me to the first interesting patent of MARCONNET of 1909, and I am of the opinion that the most prescient views as expressed therein deserve our fullest appreciation<sup>2</sup>.

When I refer in this paper exclusively to my own work, I do so because of the limited time available to me. A more detailed presentation of the history of this particular technical field would be both interesting and instructive.

When, in 1928 to 1930, I began to think about the problems of aeronautics, it was merely a sort of hobby, since at that time I was most busy as a consulting engineer in the sphere of fluid dynamics. Out of mere curiosity and interest I devoted some thought as to how small aircraft could be designed so as to be able to take off and land vertically, whilst still retaining the proven shape of conventional aircraft.

Consequently, the research was to be directed towards designing a propulsive unit of small weight which was capable of carrying the total weight of the aircraft.

Without any knowledge of earlier developments in this field, I conceived the idea that such a propulsive unit could possibly consist of a thin-walled tube with explosions being produced therein in rapid succession. With an arrangement of three such tubes, it should be possible to lift and to control an airplane.

The solution thought of is shown in Fig. 1, which accompanied my patent application of 1930<sup>3</sup>. In the lower portion of the illustration three tubes can be seen, two of which are mounted in the wings and one in the fuselage. The air-gas jets are deflected towards the ground at the exit end of the tube, thus providing three supporting jets. The calculations made proved that the space

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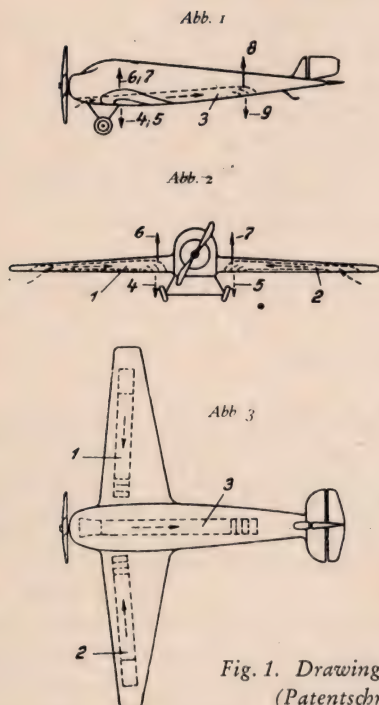


Fig. 1. Drawing from the patent application of 1930  
(Patentschrift 567 042, Kl. 62 b, Gr. 37<sub>02</sub>)

requirements and the weight of such intermittent pulse-jet engines could be kept sufficiently low, provided the combustion process could be performed with sufficient efficiency.

## 2. THE INVENTION

The pressure of the explosions produced in the tube was to supply the desired thrust and the expanded high-speed explosion gases would suck into the tube a maximum quantity of fresh air. A portion of this air would be mixed with the fuel and the remainder would serve to produce additional thrust. For this purpose it was necessary to incorporate a valve in the intake.

Fig. 2, which also accompanied a patent of that time, conveys an idea of the supposed method of operation<sup>4</sup>. The front portion of the top tube of the illustration is filled with an air-fuel mixture (illustrated by small dots). The tube immediately below shows the left hand flaps closed. The mixture has been ignited and the explosion gases convey the total contents of the tube toward the right. This is illustrated by the arrow in the rear end of the tube. The next sketch illustrates the flow within the tube after the pressure of the explosion gas has fallen. The accelerated mass continues to flow to the right, thus producing a section of low pressure in the intake end of the tube. The negative pressure causes the valve to open and fresh air is sucked in. Prior to the termination of this sucking process, fuel is added to the air to form a certain mixture. This stage of the process is shown by the lower tube of the illustration. The

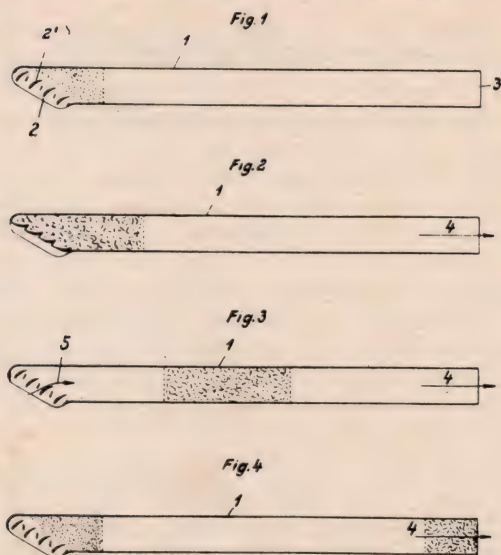


Fig. 2. Method of operation of the Schmidtrohr

subsequent closing of the valve and igniting of the mixture results in a repetition of the process.

My ideas for a powerful tubular jet engine were theoretically correct, as I was assured by a competent scientist around 1930/31. However, an apparently insignificant matter proved to constitute a major objection — it was the ignition system, for the known ignition velocities were much too slow for this process.

In order to overcome this difficulty, at the beginning of 1931 I invented a novel ignition method so as to obtain ignition at a sufficiently high rate. According to this invention, ignition was to be produced by a shock wave. If it were possible to obtain ignition, such a wave would provide an ignition velocity of several 100 m/sec, i. e. more than 10 times as fast as any known ignition velocity.

### 3. THE DEVELOPMENT

#### 3.1. The Principles of the Ignition System

Papers <sup>5</sup> on the development of the Shock-Wave Ignition Method were published by me some years ago so that I need not discuss this extensive subject in detail. Moreover other authors have recently published reports on tests of ignition produced by shock waves. In this connection I would like to refer you to the report by SHEPHERD "Third Symposium on Combustion" <sup>6</sup>, and also to the recently announced book by GREENE and TOENNIES "Chemical Reactions in Shock Waves" <sup>7</sup>.

Our practical work which began in 1931, was primarily concerned with the ignition effect of a shock from a hot gas on a cold mixture. The results of the tests could not be explained by the well known laws of the dynamics of gases

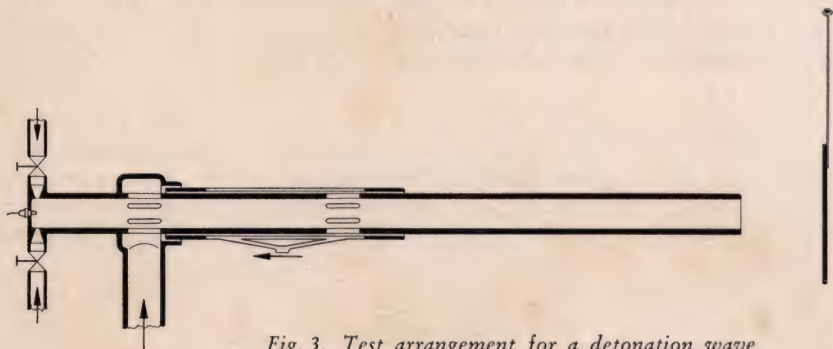
and, therefore, consideration was given to the relationships of molecular kinetics. In this way we finally obtained satisfactory agreement.

I began the development of the ignition system by augmenting my knowledge of the theoretical background, and the effects of shock waves, from the relevant literature, since up to that date I had no intimate knowledge of this particular subject. In a dissertation by R. WENDLANDT of 1923, I found my views on the ignition effect of shock waves confirmed in principle<sup>8</sup>. The shock wave — in this dissertation — was produced by exploding oxy-hydrogen gas ( $2\text{H}_2 + \text{O}_2$ ). Fuel-air mixture was not investigated by WENDLANDT.

In order to determine the effects of a detonation wave on such a mixture — this constituted the problem — I devised a test arrangement consisting of a tube of approximately 60 mm diameter. The first 250 mm of this tube were filled with an acetylene-oxygen mixture, and the second section of 500 mm length contained the fuel-air mixture. The final 1000 mm was provided with an air intake valve and was open at its rear end.

The test equipment is shown in Fig. 3. The acetylene-oxygen mixture and the fuel-air mixture were fed in under pressure, the feed being controlled periodically. Whilst the mixture was being fed in, the fuel-air and acetylene-oxygen compartments were separated by means of a controllable hinged flap (not shown in the diagram) immediately in front of the fuel-air valve.

The acetylene-oxygen gases burnt earlier were exhausted separately whilst the shut-off was in operation. As shown in Fig. 3, a baffle plate was suspended at a certain distance behind the exhaust end of the tube so that the pressure impulse of the gas jet which issued periodically could be measured.



*Fig. 3. Test arrangement for a detonation wave*

During the test, the fuel feed to the combustion air was shut off temporarily, whilst the feed of the combustion air was maintained. The modification in the operation thus consisted only of a temporary shut-off of the fuel supply. It was ascertained that the impulse of the gas jet became considerably greater as soon as fuel was fed in. Thus it was proven that the fuel-air mixture was ignited effectively.

Based on this knowledge, a systematic research on the ignition effects of shock waves produced by the explosion of compressed fuel-air mixtures was conducted from 1931 onwards. For this purpose, a device was designed to produce a single shock wave (see Fig. 4). This device consisted of an explosion chamber *a* followed by a combustion tube *b*. The explosion chamber was

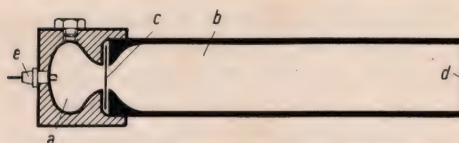


Fig. 4. Ignition device for a single shock wave

*a* = Explosion chamber  
*d* = Paper diaphragm

*b* = Combustion tube  
*e* = Spark plug

*c* = Diaphragm

separated from the tube by means of a diaphragm *c*. The tube was supplied with fuel and air at atmospherical pressure and at the ambient temperature. The exhaust closed by a paper diaphragm *d*, which prevented the mixture from escaping. Some ethyl ether or gasoline was fed into the chamber *a*, followed by compressed air at approximately 5 ata, and this mixture was then ignited by the spark plug *e*. The pressure of the explosion ruptured the diaphragm *c* and the expanding gas sent a shock wave into the tube *b*. The investigation of the combustion occurring in the tube was extended to various fuels such as gasoline, ethyl ether, hydrogen and carbon disulphide. By proper control of the volume of the explosion chamber, of the excess pressure of the explosive mixture therein, and of the tube diameter, good shock-ignitions of the mixture in the tube were obtained.

The diagram in Fig. 5 shows the pressure distribution within the tube shortly after the diaphragm *c* has been torn. The shock wave moves along the tube

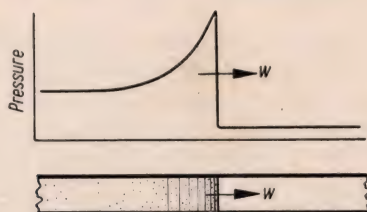


Fig. 5. Pressure distribution in tube *b* of Fig. 4

*w* = Velocity of shock wave

from left to right at the supersonic velocity *w*. The gas ahead of the wave front is not disturbed and thus is at atmospherical pressure and ambient temperature. On arrival of the shock wave, the pressure and the temperature of the gas within a narrow zone are suddenly increased. Thus combustion is initiated and the gas behind the shock wave is compressed accordingly.

The effects of a shock of cold gas were investigated in additional tests in 1932; a device as shown in Fig. 4 was used, the only modification being the absence of the spark plug. For this purpose, the chamber *a* was filled with hydrogen at ambient temperature; some nitrogen was added in order to obtain a specific weight similar to that of the combustion gases. On increasing the pressure of the hydrogen to approximately 20 ata, the diaphragm *c* was torn. Under these conditions the shock wave, acting on the mixture of carbon disulphide and air contained at atmospherical pressure and temperature in the combustion tube, produces efficient combustion. Because of the expansion, the temperature of the moving gas which produces the shock (shock gas) is lowered

to approximately  $-100^{\circ}\text{C}$ . As in the case of the "hot" shock gas, the initial pressure of the "cold" shock gas must exceed a certain minimum value in order to achieve combustion. If the pressure was below this value, no combustion could be obtained.

During these tests, the question arose as to whether this "cold-ignition" might be perhaps an electric ignition, caused by friction, interrupted spark or another reason. The energy contributions of electrical processes of this kind are, however, very small. Experience shows that they should be substantially greater in order to be able to act on the contents of the tube in the way, and with the regularity, described above, despite the very low temperature of the shock gas. Furthermore, it has been established that the additional arrangement of a bronze mesh behind the diaphragm — thus eliminating electrical tensions — does not alter the ignition effect of the shock gas. If electric effects exist, the precisely defined minimum pressure of the shock gas cannot be explained. No ignition could be observed at pressures below the minimum pressure, neither with a hot nor with a cold shock gas. Since there were no other sources of ignition present (spark plug or the like), only the action of the shock wave can be considered to be the cause of ignition.

I would like to remark that such tests later were repeated with the tube contents consisting of an ethyl ether-air mixture. The same results were obtained, with the "cold" shock gas having a pressure of approximately 75 ata excess pressure.

### 3.2. Efficiency of Combustion

At the beginning of the tests, the efficiency of the combustion was determined by an arrangement employing a suspended baffle plate (Fig. 6). A tube *c* open at one end was attached to the combustion tube *b*. A thin paper diaphragm *d* was clamped between the combustion tube and tube *c*. The air *A* within tube *c* was accelerated by the ignition of the mixture *M* in tube *b* and flowed against the baffle plate *e*. Since the plate was attached to a swinging lever, it moves to the right due to the impact of the air and, subsequently, of part of the combustion

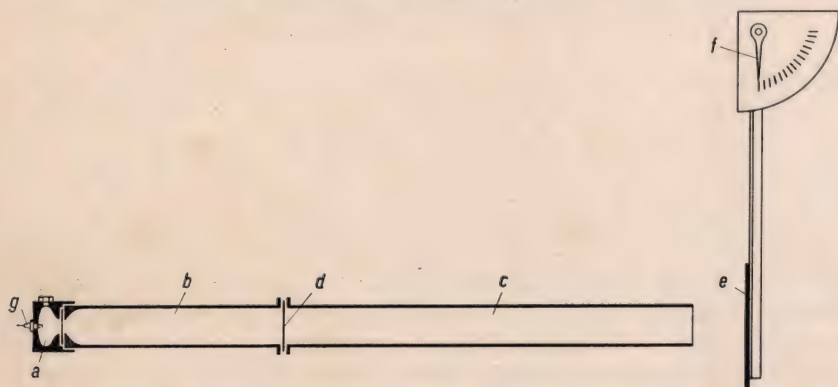


Fig. 6. Ignition device for a single shock wave with additional air

- |                              |                            |                     |
|------------------------------|----------------------------|---------------------|
| <i>a</i> = Explosion chamber | <i>b</i> = Combustion tube | <i>c</i> = Air tube |
| <i>d</i> = Paper diaphragm   | <i>e</i> = Baffle plate    | <i>f</i> = Pointer  |
| <i>g</i> = Spark plug        |                            |                     |

gases. The maximum deflection was recorded by the pointer *f*. The shock gas contained in the explosion chamber *a* was ignited by the spark plug *g*.

Mixtures of different fuels and of different degrees of saturation were tested in this test arrangement. At the same time, the influence of different quantities of air contained in tube *c* was investigated. For this purpose, tubes of different lengths, but of the same diameter, and also tubes of the same length, but of different diameters, were tested.

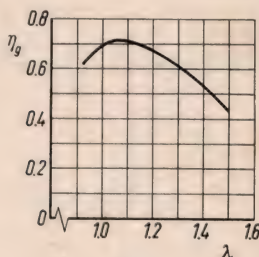


Fig. 7. Efficiency of the combustion

Fig. 7 represents the efficiency of combustion (a fraction of the energy theoretically obtainable as measured at different air-fuel ratios  $\lambda$ ). The maximum value obtained is 0.71. In this diagram, the air column contained in tube *c* is treated as a rigid body. Thus the losses incurred in accelerating the air column, are included in the efficiency. Also losses incurred (in the air tube *c*) due to advance flow occurring during the acceleration of the air in tube *c*, are taken into account.

The efficiency of combustion was determined some years later by the Engineering Laboratory of the Technische Hochschule of Dresden where, in a similar test arrangement, the gas pressure at the bottom of the tube was recorded by means of an oscillograph. These tests were initiated by Prof. Dr. A. NÄGEL and performed by Dr. S. MEURER.

In these tests, the air column was varied by means of tubes of different lengths which were attached to the combustion tube. From this arrangement, the efficiency of the combustion at different conditions was obtained in terms of the ratio  $A/M$ .

In Fig. 8 the abscissa represents the time expressed in thousandths of a second. The ordinate represents the gas pressure as measured at the bottom of the tube, for the ratios  $A/M = 0, 2$ , and  $4$ . According to the momentum theorem the duration of the pressure effect increases with the quantity of the air *A*.

In Fig. 9, the efficiency of combustion, as determined by the Dresden Engineering Laboratory, is plotted versus the ratio  $A/M$ . With  $A/M = 0$ , the air tube was omitted. The combustion gases thus expanded immediately into the open air, and probably part of the mixture left without having been ignited. This explains the fact that only a poor efficiency was obtained with the  $A/M$  ratio = 0. When an air tube is attached, the efficiency of combustion increases to a value in excess of 0.8, which confirms the previous measurements performed with a baffle plate (see Fig. 7).

The results of the determination of the efficiency of combustion by individual ignition are important in that, as will be shown later, a substantially poorer efficiency of combustion was obtained from periodic operation of the

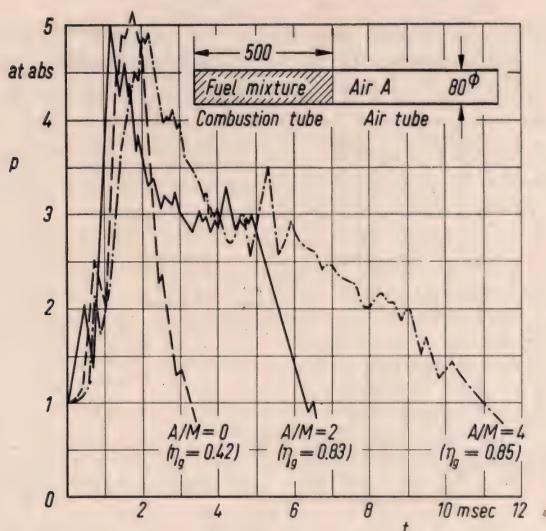


Fig. 8. Pressure in the combustion tube at different ratios  $A/M$   
 $A = \text{Air}$        $M = \text{Fuel mixture}$

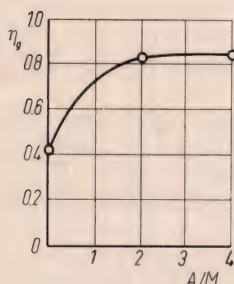


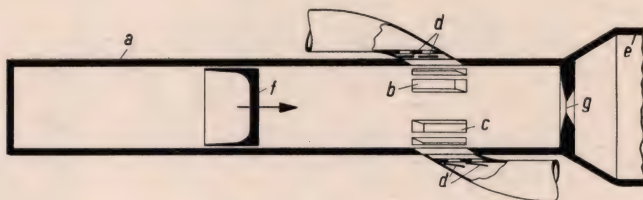
Fig. 9. Efficiency of the combustion

tube. The efficiency in stationary operation was only approximately 0.22, i. e. one fourth of the value technically obtainable. Since according to the LENOIR-Process, combustion theoretically supplies only approximately 30 per cent of the chemical energy in the form of mechanical energy, it is very important in technical applications to know whether this 30 per cent is used with an efficiency of 0.8 or of only 0.22. Furthermore, it is important that, when a tube containing air is attached, this air is accelerated without incurring greater losses. Thus these test results form a considerable part of the foundation required in the evaluation of the technical usefulness of the principle.

### 3.3. Operation with an Ignition Device

Towards the end of 1934, an ignition device had been developed which supplied shock waves periodically at 50 c/sec. This device is shown in Fig. 10. Check valves  $d$  were attached to the cylinder  $a$  ahead of the slots  $b$  and behind the slots  $c$ . These valves admitted the mixture into the cylinder via the slots  $b$ , whilst the excess mixture not utilized in combustion left the cylinder via the slot  $c$ . The mixture was then fed into the combustion tube  $e$ , the end of which

is indicated (the valve is not included in the drawing). As it advanced, the free piston *f* at first pushed the mixture not used for ignition out of the cylinder via the slots *c*, and then compressed adiabatically the mixture enclosed between the slots and the bottom of the cylinder until self-ignition occurred. A shock



*Fig. 10. Ignition device*

<i>a</i> = Cylinder	<i>b, c</i> = Slots	<i>d</i> = Non-return flaps
<i>e</i> = Combustion tube	<i>f</i> = Piston	<i>g</i> = Aperture

wave left via the aperture *g* provided in the cylinder bottom and ignited the mixture contained in the combustion tube. The gas still remaining in the cylinder imparted to the piston the required backward motion of approximately 70 to 80 m/sec. On the return travel, the piston sucked in fresh mixture via the slots *b*. At the same time, the piston's velocity was reduced by the compression of the air enclosed behind the piston. The compressed air gave a forward motion to the piston and thus the cycle was repeated. In order to make sure that the rear end of the cylinder always remained filled with air only, fresh air was introduced continuously, at slight pressure, via a device not shown in the drawing.

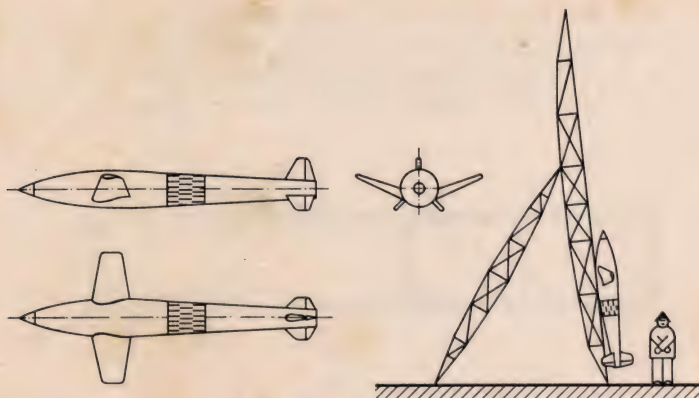
For several years various modifications of this type of ignition device were operated successfully.

Since it was the primary object to obtain fundamental knowledge on the most suitable tube dimensions, the thrust obtainable and the fuel consumption, only short duration tests were run. Other tests performed with ignition systems extended to simple combustion tubes, and to combinations of combustion tubes and separate air-filled tubes, by means of which considerable quantities of air were accelerated by the combustion gases.

### 3.4. Influence of the German Air Ministry

In 1934 it had become somewhat difficult to raise the financial means required for the further development which constituted but a side-line of my normal occupation. I tried, though without decisive success, to interest and to obtain assistance from, outside circles for the idea of the vertical take-off and landing. Following the advice of a friend, I also worked out in 1934 some applications of the pulse-jet engine for defense purposes. Here I was assisted by Prof. Dr. G. MADELUNG, who worked out the aerodynamic data for a flying bomb, an interceptor fighter and a light bomber. The dimensions of the pulse-jet engines were based upon the trial results which I had obtained with individual ignition.

I submitted this worked-out project as a Memorandum to the Reich Air Ministry then in course of being established. Fig. 11 shows the proposal, from



*Fig. 11. Project of the flying bomb*

this memorandum, for a flying bomb, which was later called the V-1. It will be seen that in this proposal the pulse-jet engine was fixed as an elongation at the rear of the fuselage. The speed of this body was calculated as 800 km/h at an altitude of 2 km. However, the proposals of this memorandum were not appreciated. At that time they were refused as being technically dubious and as uninteresting from a tactical point of view.

Nevertheless, in 1935 my being sponsored by Prof. A. NÄGEL, Dr. LORENZ, Prof. BUSEMANN, Dr. DORNBERGER and Dr. VON BRAUN resulted in the Reich Air Ministry and the Army Ordnance Office taking an active interest in the matter by granting financial funds which had to be paid back at a later date. I was able to re-pay these funds by the end of 1944. They amounted to somewhat less than one million Mark.

Thus, from 1935 onwards development progressed more speedily than before. Soon we had tubes operating, equipped with ignition devices and making it possible to establish data on thrust and consumption during periodic running.

### 3.5. Automatic Ignition

In 1937 the first oscillographic measurements were made and we found out that the ignition system operated at 50 c/sec, whilst the tube tested at that time operated at 100 c/sec. Thus every second ignition was initiated automatically. We were not too surprised about this discovery, for during the previous tests we occasionally established by aural perception that one or the other tube continued to operate 3 or 4 periods after the ignition system had been switched off. In most of the tests, approximately 1000, however, the engine stopped immediately the ignition was switched off.

We decided that the automatic ignition was due to the action of a weak shock wave travelling from the end of the tube back into the tube after the explosion took place and thus caused the second ignition. We intended to increase this shock wave in order to obtain automatic ignition. For this purpose we increased the resistance of the inlet valve at the intake end of the tube a little, and the success was striking. The engine immediately operated with automatic ignition and the ignition device previously necessary was no longer required.

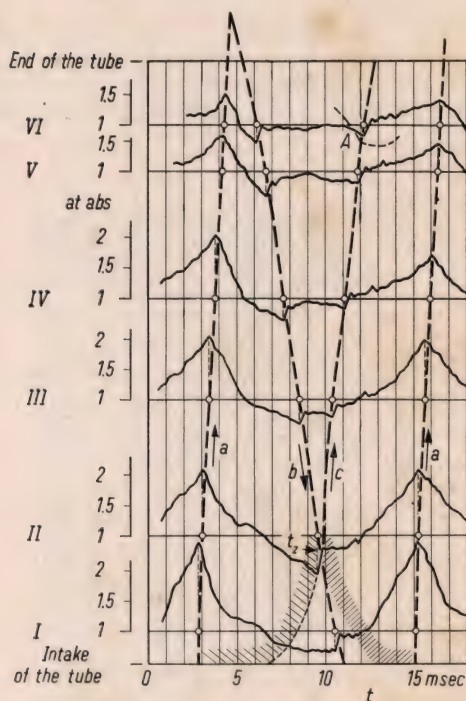


Fig. 12. Pressure oscillograms

The oscillograms of the pressure as obtained from this tube in 1938 are shown in Fig. 12. The pressure distribution curves are plotted versus time for various tube diameters. The dashed line shows the distribution of the maximum pressure and of the shock wave returning from the end of the tube. It was a cylindrical tube about 2 m long, the diameter was 120 mm, and it was provided with check valves. It was operated with an ether-air mixture, which was sucked in, by the tube, from a tank. The ignition effect of the shock wave having a pressure jump of only about 0.3 atm. is surprising. According to our tests, a minimum pressure jump of approximately 0.3 atm. is required in periodic operation for ignition of cold fuel mixture by shock waves coming from the combustion gas. The size of the pressure jump required depends on the temperature of the mixture, thus ignition of a pre-heated fuel mixture can be obtained with a still smaller pressure jump.

The evaluation of the test data shown in Fig. 12 makes — by using further measuring data and experience — a qualitative determination of the flow within the tube possible. This is shown in Fig. 13. Besides the pressure curves *a*, *b*, and *c* of Fig. 12, the flow curves are plotted versus the time between which equal proportions by weight of fuel mixture, gas or air are included. Flow curves delimiting a new charge against the gas left over from the previous combustion are referred to as *d*. The new combustion gases fill almost the whole tube, but still do not flow out.

For determining the condition of the gas, it is assumed that minor irregularities approaching the velocity of sound *a* are present. From the

oscillograms it is possible to determine the rate of travel, and thus the gas temperature can be calculated for a small tube length — at times and in sections of the tube where no combustion occurs, and where there is only negligible gas motion present. For the purpose of Fig. 13, three different gas

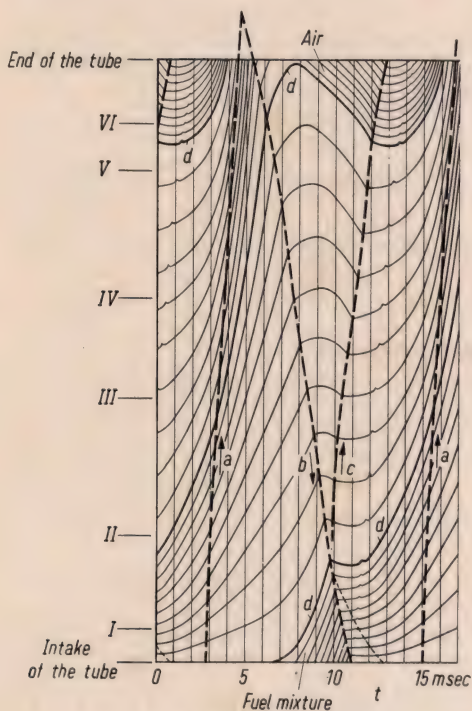


Fig. 13. Diagram of flow

*a, b, c = Pressure curves of Fig. 12      d = Flow curves delimiting new charge*

pressures were taken. They show — when reduced to the same pressure — a suitable correspondence. A further calculation was performed, in which it was assumed that the processes follow the adiabatic law sufficiently closely. Since the influence of the shock waves on the flow is relatively small, an estimate of the course of flow within the range of the waves could be obtained from the momentum theorem.

More powerful tubes which normally contain a bigger charge and which have more efficient combustion, show, in stationary operation, an expansion of the combustion gases beyond the end of the tube. The flow characteristics, however, are rather similar.

The most important technical feature found in the tests is the fact that, in the case of stationary operation, the quantity of fuel mixture that may be fed into the tube during one cycle has an upper limit. If a greater quantity of mixture is added, no automatic ignition will result. The practical solution of this problem was to provide the tube inlet with a certain minimum resistance.

### 3.6. Thrust Transfer

Periodic combustion involves a gradual power production resulting in power peaks, which exceed the average manifold thrust. In order to compensate the power peaks, the tube was connected to a mounting by means of a spiral spring. Thus the tube behaved as a vibrating mass, excited by the pressure pulsation of the combustion gases. With this arrangement, a power fluctuation of only about 1% of the average thrust is obtained at the point of fixture. The tube compresses the spring by about 10 to 20 mm, and in this position the tube oscillates through some tenths of a millimetre. Only the change in the travel of the spring is transferred as power fluctuation to the point of fixture of the tube.

The influence of the parameters of the spring on the levelling of the power fluctuations for a tube developing a thrust of 500 kg is shown in Fig. 14. The curve represents the maximum value of the force; the average thrust is represented by the dashed line. The abscissa represents the order of magnitude of the spring parameters  $c$ . The design chosen is marked by a circle.

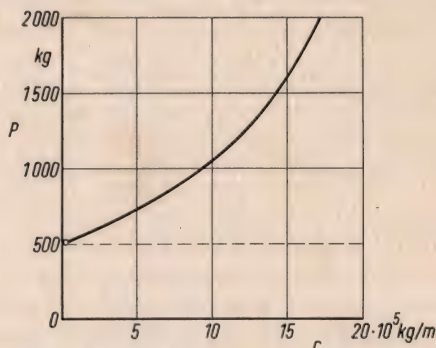


Fig. 14. Maximum force  $P$  at the point of mixture of the tube depending on spring parameter  $c$

During operation, the amplitude of the force at the point of fixture is equal to the value by which the full curve exceeds the dashed line. When the maximum value approaches 1000 kg, the minimum value tends to zero. If a still greater spring parameter is chosen, negative power peaks will occur in addition to the positive ones. As the spring parameter increases, i. e. in the case of less resilient suspension, the vibration of the tube approaches the condition of resonance, in which powerful positive and negative forces occur.

For test purposes, in addition to a resilient suspension of the tube a hydraulic damping system was provided. The hydraulic system was composed of an oil-filled cylinder and a piston having adjustable leakage. Sudden changes in the operating conditions were thus attenuated aperiodically.

The system of resilient suspension and attenuation of the tube was applied successfully in our tests. In no instance were troublesome influences that would have been caused by the pulsating operation of the tube observed.

### 3.7. Tube with Additional Air

Up to 1938 we were primarily concerned with the development of the combustion tube, since this work had to provide the basis for any further

development. In stationary operation, this combustion tube sucks in air at the end of the tube after every combustion. This results in additional air in stationary operation, which is eliminated, however, at higher air speeds.

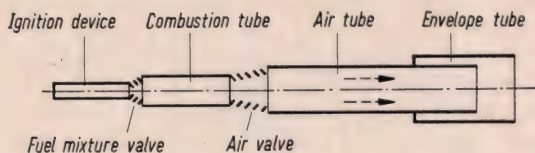


Fig. 15. Tube with additional air

In order to obtain arbitrarily a greater quantity of additional air, systems with separate air tubes were also tested up to 1938. Fig. 15 shows one of these test arrangements. The combustion tube, to which an air valve with a larger air tube was added, was attached to the 50 c/sec ignition device (left-hand-side of diagram). The end of the air tube was provided with an envelope tube, with its end closed by an annular bottom. It was intended to favourably influence the natural vibrations of the air column within the air tube through the natural vibrations in the envelope tube; however, we had to discontinue this promising project. In 1938 it was decided to stop the development of additional air (to be chosen arbitrarily) and to use all our means for the further development of the simple combustion tube.

### 3.8. The Development of the Valves

Therefore, from 1938 onwards, only the further development of the simple combustion tube was continued. The first effort was spent on the design of valves that could be used in practical operation. After having tested various valve forms, flaps and controlled or freely hinged valves, we selected for the purpose of further development a check valve. In the beginning of 1941 we obtained, with a certain design, a life of more than 20 hrs. of operation of the tube with a specific thrust of about 0.28 kg/cm<sup>2</sup>. It was not our intention to retain valves with normal non-return flaps. But it was not possible to carry out further developments of this.

Substantial progress was made in the design of valve flaps, after it was realized theoretically in 1938 that the natural frequency of the flap should be chosen in a certain ratio to the frequency of the exciting force. When a non-

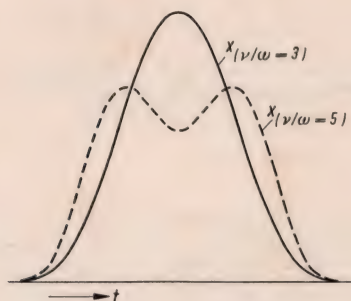


Fig. 16. Non-return flap

return flap (see Fig. 16) is pushed into the open position by a harmonically varying excess pressure then, due to the stiffness and mass of the flap, the displacement  $x$  of the end of the flap is given by:

$$x = \frac{x_{st}}{1 - (\omega/\nu)^2} \left( \sin \omega t - \frac{\omega}{\nu} \sin \nu t \right).$$

In this equation  $x_{st}$  is the static flap displacement due to the excess pressure,  $\omega$  is the frequency of the exciting force,  $\nu$  is the natural frequency of the flap and  $t$  is the time. From this equation it follows that if  $\nu/\omega$  is odd ( $\nu/\omega \geq 3$ ), the flap returns to its original position with zero velocity. Thus the flap is prevented from striking too hard against its seat, which otherwise would destroy the valve in a matter of seconds.



*Fig. 17. Diagram of flap deflection*  
 $t$  = Time  
 $\omega$  = Frequency of exciting force  
 $\nu$  = Natural frequency of flap

The full line in Fig. 17 represents the flap deflection for  $\nu/\omega = 3$ , and the dashed curve represents this deflection for  $\nu/\omega = 5$ . Normally, these theoretical conditions are influenced by the assembly of the flaps. In particular the pressure drop of the air within the flap channel plays an important part. Moreover, the theoretical relationship provides for a pre-opening of the flap, i. e. a position of the flap free of stress, thus providing a certain opening of the valve. In periodic tube operation, a substantial increase in the quantity of air flowing through follows from such a positioning of the flap. This arrangement permits the assembly of a valve requiring only a little space.

From 1938 to 1940 larger tubes were also constructed and tested; up to 1938 we had tested only 120 mm tubes. In 1939 we increased the tube end diameter to 200 and then to 510 mm. Those tubes were approximately 3.5 metres long. The first 510 mm diameter tube operated in the beginning of 1940 and supplied a thrust of 500 kg in stationary operation.

### 3.9. The Preparation of the Fuel Mixture

A particularly simple kind of mixture preparation was tested in a 200 mm diameter tube. Fig. 18 shows the tube  $a$  with the inlet valve  $b$ . In the tubular inlet piece  $c$  there is an annular tube  $d$  containing some nozzles  $e$ . Liquid fuel is supplied continuously at a pressure of some atmospheres to the annular tube  $d$  via the pipe  $f$  so that the nozzles supply a continuous fuel spray into the tubular piece  $c$ . The axial velocity of the fuel spray is adapted to the periodic velocity  $g$ . The tubular piece  $c$  is relatively short and the periodic velocity  $g$  is not so great that important air oscillations occur. The air flows intermittently through the tubular piece  $c$  in the direction shown. At the instant of the explosion of the mixture in the combustion tube, the air in the tubular piece  $c$  is thus practically at rest, and during this time interval, the fuel spray from the nozzle  $e$  proceeds into a region whose volume corresponds to that sucked in by the tube during each period. Thus the air is provided with about half the quantity of fuel required. During the subsequent suction stroke, the pre-saturated air

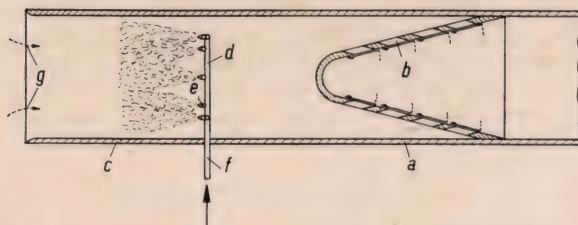


Fig. 18. Tube with mixture preparation outside the combustion tube

- |                         |                        |                       |
|-------------------------|------------------------|-----------------------|
| <i>a</i> = Tube         | <i>b</i> = Inlet valve | <i>c</i> = Inlet tube |
| <i>d</i> = Annular tube | <i>e</i> = Nozzles     | <i>f</i> = Fuel pipe  |
| <i>g</i> = Air velocity |                        |                       |

then passes the nozzles *e* and thus is provided with that portion of fuel it lacks for the proper preparation of the desired mixture. There will always be some prepared mixture in that portion of the tubular piece between the nozzle *e* and the valve *b*.

The mixture preparation with continuous feed of the fuel operated very well. The pressure oscillograms show very regular operation of the tube. However, this method of preparing the mixture was not continued since it took place outside the tube and thus entailed the risk of fire.

For this reason, the development of a periodic mixture preparation within the tube was started in 1939. In the spring of 1940 the so-called atomizers were developed (shown in Fig. 19). The atomizers are arranged closely behind the

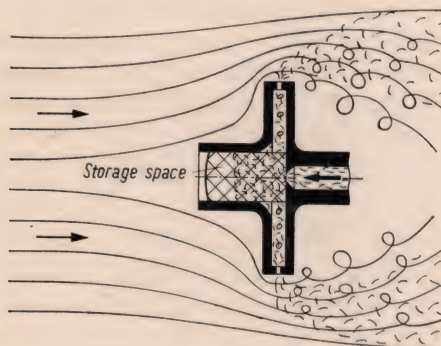


Fig. 19. Atomizer

valve, i. e. in the front portion of the combustion chamber. A steady flow of fuel enters the atomizers — in the diagram coming from the right-hand side — its flow energy is eliminated in the storage portion that is filled with a wire mesh. The storage portion is open towards the valve so that the pressure head formed by the periodic air intake can act on the surface of the fuel. Holes were machined into the rim of the hollow atomizer plate, through which an amount of fuel corresponding to the pressure head is ejected. This way we obtained an automatic and periodical admixing of fuel into the combustion air.

A 510 mm diameter tube developing a thrust of 550 kg was operated with such atomizers before the year 1940 was over. Fig. 20 represents an oscillogram of the thrust, gas pressure and fuel consumption. In terms of the maximum diameter of the tube, the specific thrust was  $0.27 \text{ kg/cm}^2$ . The fuel consumption per 1 kg of thrust and 1 hour was 2.8 kg.

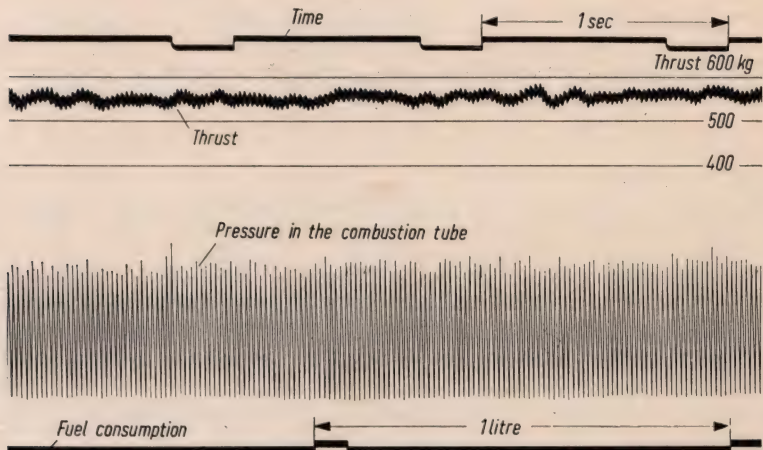


Fig. 20. Oscillogram of the 510 mm diameter Schmidtrohr

Fig. 21 shows the design of the tube. On the left-hand side there is the valve for the combustion air; the valve is composed of tapered air ducts and flaps. We also tested with equal success tubes of similar sizes which were provided with flap valves arranged normal to the tube. The diameter of the tube is reduced to 450 mm close to the valve, and the atomizers are arranged in this position. A tapered tube then connected this to the cylindrical portion of the tube of 510 mm diameter.

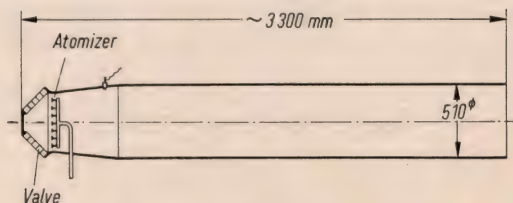


Fig. 21. 510 mm diameter Schmidtrohr

In the development of the propulsive unit, we endeavoured to satisfy the condition that the diameter should always be less than the end diameter of the tube. This demand arose because of the aerodynamic conditions prevailing when a tube is attached to a flying body. It became evident that it is necessary to feed the fuel at the frequency of the air intake in order to meet the above requirement. Otherwise an enlarged combustion chamber has to be provided, whose cross-section is approximately twice as great as that of the end of the tube. Since the size of the cross-section of the tube end is precisely proportional to the thrust

produced, the extension of the diameter beyond the tube end diameter results in substantial additional resistance. It seemed important to us that this be avoided by proper design.

#### 4. INTERDICTION OF DEVELOPMENT

At this state of development in spring 1941 I had the impression that, during 1940, about half a year had been lost in the development of the tube. My responsibility for this subject induced me to report accordingly to the Ministry.

Shortly afterwards, the authorities interdicted further research work regarding the tube to be carried out by my firm.

Regarding the loss of time, however, I continued to report to the Ministry; at the beginning of 1944 I estimated the loss of time as amounting to  $2\frac{1}{2}$  years.

Some years ago I had started, on a minor scale, work on the development of a propulsive unit with high pre-compression of the mixture. The impulse for this work was given by the experience gained with the afore-mentioned ignition device. Since, with this device, the freely moving piston attained velocities up to 80 m/sec and high compression of the mixture, it seemed quite obvious to develop this construction for the combustion of all the propelling gas. With the assistance of the Air Ministry I was able to continue this work.

Furthermore, following the interdiction of development work on the simple combustion tube, my firm received an order to manufacture several tubes showing the state of development reached by us at that time. In connection with this work I occasionally found an opportunity to test minor improvements of the tube.

Thus, in 1942 I had tuned up a 500 mm diameter tube by conical enlargement to 750 kg stationary thrust, because I had heard that it was intended for taking-off acceleration. This tube had 450 mm diameter at the front end of the combustion chamber, 565 mm diameter at the end and a total length of about 3.5 m. Specific consumption in stationary operation amounted to 2.75 kg fuel per kg thrust and hour. Specific thrust in stationary operation amounted to 0.3 kg/cm<sup>2</sup> in relation to the maximum cross-section of the tube.

#### 5. AUXILIARY EQUIPMENT

##### 5.1. Starter

For starting larger tubes, a starter was developed in order to ensure starting at a specific moment. A 24 V electric motor was used for the short-time operation of an axial-flow blower unit, whose air stream was provided with fuel, and the mixture then was fed into the combustion tube from the centre of the valve. Fig. 22 shows the electric motor *a* (left-hand side) coupled to the impeller *b*. Six hinged valves *c* are arranged between the impeller and the tube front end; in Fig. 22 they are shown in axial position. When the motor is being started, they are turned automatically in this direction. When the motor is being switched off, they turned automatically for 90° and shut the short feed pipe *d*, thus sealing the combustion tube against the atmosphere. A fuel nozzle *e*

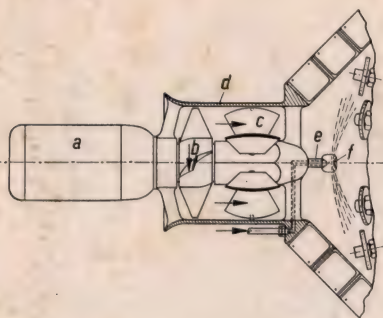


Fig. 22. Starter

- a* = Electric motor
- b* = Impeller
- c* = Hinged valves
- d* = Feed pipe
- e* = Fuel nozzle
- f* = Baffle plate

together with the fuel pipe is incorporated in the end of the short feed pipe *d*. The nozzle *e* sprays fuel against the baffle plate *f* so that the fuel is mixed into the air stream of the blower. The right-hand section of Fig. 22 shows the initial section of the valve of the combustion tube and illustrates the position of the atomizers arranged in that section of the combustion tube.

A relay is arranged beside the starter blower, which is operated by the fuel when the fuel flows into the tube from a pressure tank on opening the fuel supply main valve. The relay at first turns on the motor *a* and at the same time the supply of the starter fuel to the nozzle *e*. Subsequently, the relay operates the starter spark plug in the combustion tube and the feeding of the fuel to the atomizers of the combustion tube. Immediately after the first ignition, the relay automatically cuts off the motor and the fuel supply to the blower unit.

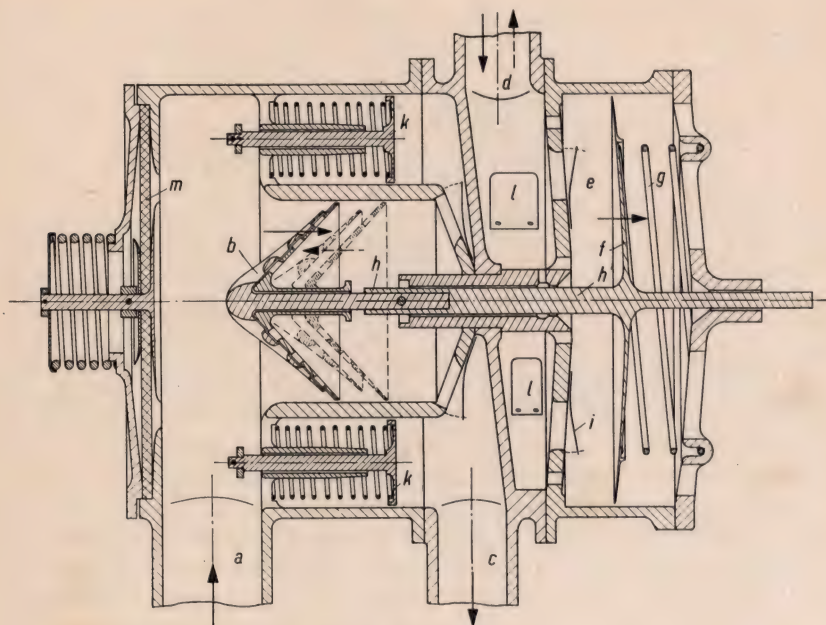
With this equipment starting was performed within 1.5 sec. The deviations to this period were less than 0.1 sec.

## 5.2. Fuel Pump

For delivering fuel from the tank, a pump was developed that was operated by the pressure pulsations of a tube of approximately 70 c/sec. The design of that pump is illustrated in Fig. 23. The fuel is fed via the short feed pipe *a*, it is conveyed by the piston *b* and it is discharged via the outlet pipe *c*. The piston *b* is designed as a suction valve. The pump is operated by the pressure pulses that arrive in the cylinder *e* from the combustion tube via the pipe *d*. The piston *f*, loaded with a pressure spring *g*, operates within the cylinder *e*. The piston *f* is connected to the piston *b* via the shaft *h*.

When, at the moment of combustion in the combustion tube, an excess pressure occurs in the pipe *d*, this pressure enters the cylinder *e* via the valve flaps *i*, and thus moves the piston *f* against the pressure spring *g*. At the same time the fuel is fed into the pipe *c* by the piston *b*. In order to obtain a uniform discharge, spring-loaded equalizer pistons *k* are provided, which allow part of the fuel to enter the cylinder sections.

If there is a negative pressure in the pipe *d*, the piston *f* will return. The flaps *i* are closed, however, so the pressure gas can escape via the holes provided in the bottom of the cylinder *e*. Non-return flaps *l* are arranged between the cylinder *e* and the pipe *d*, and provide a connection with the atmosphere in the case of a negative pressure in the pipe *d*. Thus the air enters the pipe *d* and, due to the leakiness in the bearing of the shaft *h*, the air carries the fuel with it.



*Fig. 23. Fuel pump*

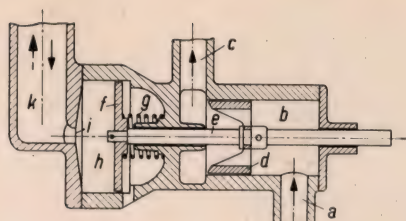
- |                              |                             |                        |
|------------------------------|-----------------------------|------------------------|
| <i>a</i> = Feed pipe         | <i>b</i> = Piston           | <i>c</i> = Outlet pipe |
| <i>d</i> = Connecting pipe   | <i>e</i> = Cylinder         | <i>f</i> = Piston      |
| <i>g</i> = Spring            | <i>h</i> = Shaft            | <i>i</i> = Valve flaps |
| <i>k</i> = Equalizer pistons | <i>l</i> = Non-return flaps | <i>m</i> = Diaphragm   |

Moreover, the periodic air supply provides good cooling of the equipment. When the return motion as indicated by the dashed line occurs, the piston *b* acts as a valve. The way in which a movable portion of the piston is detached from its solid seat and thus allows a flow through the piston *b*, is illustrated by means of dashed lines. In the top dead centre of the travel, the piston *b* effects sealing by the movable part resting against the supporting body. The diaphragm *m* is adjusted by means of a spring in such a way that it ensures a uniform fuel supply through the short feed pipe *a*, in spite of the periodic delivery.

The pump delivery was approximately 1 litre of fuel per second at a pressure of about 1.5 kg/cm<sup>2</sup>.

### 5.3. Pilot Valve

As a rule, in our tests we used an automatic pilot valve as illustrated in Fig. 24, in order to shut off the fuel supply in the case of an unexpected interruption of the operation of the engine. When the pilot valve is open, the fuel enters the cylinder *b* via the short feed pipe *a*, and it leaves the cylinder via the outlet pipe *c*. A tubular piece *d* slides in the cylinder, the motion being imparted to it by the shaft *e*. A piston *f* is attached to the left end of the shaft *e*, with a spring *g* operating on that piston. The cylinder *b* is connected with the tubular socket *k* via a small hole *i*. The tubular socket *k* opens out into the



- a* = Feed pipe
- b* = Cylinder
- c* = Outlet pipe
- d* = Tubular slide piece
- e* = Shaft
- f* = Piston
- g* = Spring
- h* = Cylinder
- i* = Connecting hole
- k* = Tubular socket

Fig. 24. Pilot valve

combustion tube so that the gas pulsations operate in the tubular socket *k*. The moving masses and the spring *g* are balanced in such a manner that the fuel supply will not be shut off as long as combustion in the tube is regular. Should combustion fail, however, the valve is closed after 2 or 3 normal cycles so that the tube would not be flooded with fuel.

## 6. A TEST IN THE WIND TUNNEL

At the turn of 1942/43 the German Air Ministry asked me to send the 750 kg thrust unit for investigation in the wind tunnel at Brunswick.

In an initial test it was found out that the delivery of the fuel pump provided at Brunswick was too small for our tube. After a second fuel pump was installed, the tube began to operate on about  $\frac{2}{3}$  of the normal fuel quantity. The engine ran rather noisily, since it operated close to the lower limit of the possible range of operation. However, the thrust could be measured, and the indicated thrust was about 375 kg at an air speed of about 350 km/h.

During this operation the building was heavily shaken and a rather excited man came to the test-stand, wanting to stop running the tube. In a side-room something seemed not to be alright; it was said that the building was in danger of collapsing.

With twice the thrust of this tube we had not experienced such shocks at our test-stand at Munich. It must be said, however, that we had fixed the tube always softly sprung, so that at the point of fixture the power fluctuation did not exceed 1% of the thrust of 750 kg. This soft springing does not appear to have been provided for in the wind tunnel arrangement. In this case it was quite possible that shocks of several tons at 50 c/sec acted upon the building.

As far as I remember, following this incident no further trials were conducted with our tubes in the wind tunnel.

The approximate indicated thrust as determined by taking into account the resistance of the tube, was recorded correctly, as can be gathered from Fig. 25.

The top curve represents the variation of the maximum thrust with air speed, as determined by computation. It declines from 750 kg of stationary thrust to 630 kg at an air speed of 350 km/h, and later it increases again. In this operation, the tube consumes the fuel quantity indicated by the full line in the bottom of the diagram.

The point of 375 kg indicated thrust as measured is marked on the dashed thrust characteristic. The corresponding point of fuel consumption is shown on

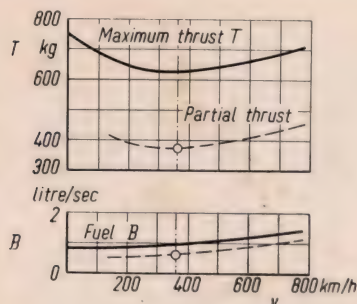


Fig. 25. Thrust and fuel consumption at different air speeds

the fuel consumption characteristic in the lower portion of the diagram. It thus becomes evident that these corresponding values are proportional to those valid for maximum thrust.

My collaborator HANS LEMBCKE established the following theoretical relationship between the thrust  $F$  in flight and the stationary thrust  $F_0$ :

$$\frac{F}{F_0} = \frac{m}{m_0} \frac{w - v}{w_0},$$

where

$$\frac{m}{m_0} = \frac{G(1+k)}{G_0(1+k_0)}$$

indicates the ratio of the accelerated masses and where

$$w = \sqrt{\frac{\eta_u}{1 + \zeta_w} \left[ \frac{2Hg}{(1 + \lambda A)(1 + k)} \eta_{th} \eta_g + v^2(1 - \zeta_v) \right]},$$

$$w_0 = \sqrt{\frac{\eta_u}{1 + \zeta_w} \frac{2Hg}{(1 + \lambda A)(1 + k_0)} \eta_{th} \eta_{g_0}}$$

are the values of the relative velocities as produced by the calorific value of the fuel.  $G/G_0$  is the ratio of the fuel mixtures combusted.  $k$  denotes the ratio of the additional air accelerated to the quantity of mixture (corresponding to the value  $A/M$  of Figs. 8 and 9).  $\eta_u$  denotes the efficiency resulting from the irregularity of the flow. With reference to Fig. 26

$$\eta_u = \frac{\int_0^{T_0} \left( 1 + a \sin \frac{2\pi}{T_0} t \right)^2 dt}{\int_0^{T_0} \left( 1 + a \sin \frac{2\pi}{T_0} t \right)^3 dt \int_0^{T_0} \left( 1 + a \sin \frac{2\pi}{T_0} t \right) dt}$$

the integration being possible only over the outflow direction of the velocities, since the temporarily opposite speeds at the tube end are balanced according to the initial equation by an equal value of the outflow in first approximation.  $\zeta_w$  and  $\zeta_v$  introduce the coefficients of the resistance to the flow through the

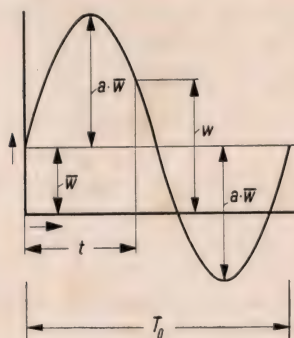


Fig. 26. Oscillation of the velocity

$t$  = Time

$w$  = Instantaneous velocity

$\bar{w}$  = Mean velocity

tube and the intake into the tube. The quantity of air theoretically required for combustion is referred to as  $A$ , i. e. the ratio of the weight of the air to the weight of the fuel; as a rule,  $A = 15$ .  $\lambda$  denotes the air,  $\lambda > 1$  being a measure of the excess air. As a rule,  $\lambda = 1.1$ . The air speed is referred to as  $v$ , the acceleration due to gravity as  $g$ . The theoretical efficiency of the combustion,  $\eta_{th}$ , is the efficiency of the LENOIR-Process, which in stationary operation is 0.3. The energy decreases with the efficiency of combustion, referred to as  $\eta_g$ .

Besides determining the thrust in the case of air flow, the above equation assisted us in particular in establishing the efficiency of combustion. We found out early that the efficiency of combustion is only 0.22 in stationary operation. We furthermore found out that this efficiency increases in a certain proportion with the increase of the combustion air flowing in under pressure. Thus an efficiency of 0.5 is obtained at an air speed of 600 km/h.

By theoretical studies it was established that the additional air column expressed by  $k$  becomes zero at air speeds between 600 and 700 km/h due to the air return at the tube end. It declines according to the equation

$$k = k_0 \frac{G_0}{G} \left( 1 - \sqrt{\frac{q}{p_0}} \right)$$

$q$  being the ram pressure and  $p_0$  the negative pressure as prevailing within the end section of the tube during the sucking-in cycle.

It is remarkable that theory shows that the performance of a tube can be determined at small air velocities. Thus the correctness of theoretically established relationships can be determined by tests where the velocity in blower stream, to which the tube is exposed, is relatively small. The results so obtained were confirmed later when the velocity in blower stream was great.

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## DISCUSSION

Prof. Dr. QUICK (Aachen): You have spoken of the ignition and in particular of the shock-wave ignition. Is it really quite clear in which way the shock process is initiated? No doubt the shock-wave ignition offers one possibility; another possibility might be, however, that the gases remaining in the tube initiate ignition. Do you know, in more detail, which of these two processes might be the more likely one?

Dipl.-Ing. PAUL SCHMIDT: I am under the impression that there are two characteristic methods of short-time ignition of mixtures.

One method consists of the well-known ignition by heat conduction with whirling of the mixture. In this way considerably shorter ignition times are obtained than by heat conduction in an undisturbed mixture. The factor of time-gain amounts to 1:10 whilst the absolute value of the ignition velocity reaches 20 to 30 m/sec.

The other method is that of shock-wave ignition. The conditions prevailing here are perhaps best explained by the working of a tube the construction of which is similar to the tube shown in Fig. 21. The tube has a length of about 3.5 m. At the entry the same valve, as shown in Fig. 21, is fitted. Directly following the valve is the combustion chamber of 0.45 m diameter. From here onwards the tube increases steadily in diameter with a conical angle of  $2^\circ$ , the diameter at the tube-end being 0.565 m. The air sucked in by the valve and mixed with fuel flows at about 250 m/sec through the combustion chamber the cross-section of which is completely filled. From the measured thrust of 750 kg, the fuel consumption and from the under pressure at the end of the intake cycle it can be computed that about 1.5 m of the tube length are filled by the mixture. This mixture has a volume of 250 litres. The time of one periodic cycle amounts to  $20 \times 10^{-3}$  sec. From the increase of pressure it follows that ignition will be terminated after about  $3 \times 10^{-3}$  sec. Thus an ignition velocity of about 500 m/sec for the mixture column of 1.5 m length is obtained.

Occasionally, the tube has been operated in the open air at  $-20^\circ\text{C}$ , and it started without lapse of time and worked through without interruption. Peak pressures of almost 5 ata were measured, so that the theoretical value can be assumed to be about 9 ata. Thus this process is to a large extent a constant-volume combustion. With a tube diameter of 0.5 m and with the forcibly axial

flow of the mixture which fills out the total cross-section, wall-influences and remaining gases may be left out of consideration. For the periodical operation of this tube there seems to me to be no feasible explanation other than ignition by shock-wave. According to Figs. 12 and 13 the shock-wave originates at the tube end when the combustion gases flow out, and from there runs backwards into the tube towards the mixture.

The same ignition velocity of about 500 m/sec is obtained also from tests with individual ignitions of an undisturbed mixture according to Fig. 8. Here the mixture column of 0.5 m length is ignited by a suddenly generated shock of hot combustion gases. These gases enter through an opening of about 10 mm diameter which is situated in the middle of the bottom of the combustion chamber of 80 mm diameter. The shock gas is generated by a device similar to the one shown in Fig. 10, though only one gas shock is generated. It appears from Fig. 8 that the maximum pressure of 5 ata is being obtained in a little over  $1 \times 10^{-3}$  sec. Thus, combustion will be practically terminated within this period. The ignition must therefore pass through the mixture at about 500 m/sec. This appears to me to be feasible only by shock-wave ignition.

In this case, the ignition effect cannot be theoretically explained by the well-known gas-dynamical laws. According to these laws the rather weak waves, which effect ignition, result in too small a temperature increase, which is much below the ignition temperature of the mixture. The shock-wave ignition may be explained, however, when the processes within the shock-front are considered from a molecular-kinetic point of view. I may refer to the British patent Nr. 737,555, where at a later date I have laid down some details on this subject.

# DEVELOPMENT OF THE V-1 PULSE JET

FRITZ GOSSLAU \*

## 1. TASK

In 1939 the German Air Ministry decided to have jet engines developed. Each of the German aero-engine factories was asked to work on different technical solutions of this task. The ARGUS MOTOREN GESELLSCHAFT, Berlin, were asked to develop a pulse jet. Curiously enough, this task was formulated as follows:

"Take a test tube, put in some drops of gasoline, shake the tube and ignite its open end. The mixture will not burn continuously, but in rhythmic pulses."

As an oscillation process of the working gases had evidently to take place, we opposed two oscillation chambers in our first model (Fig. 1).

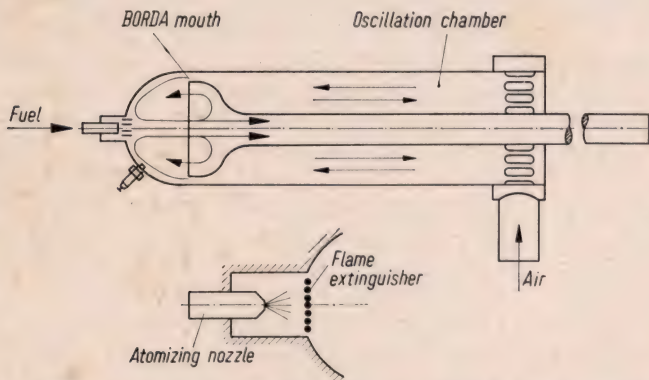


Fig. 1. Pulse jet, first Argus model with oscillation chamber. Borda mouth and flame extinguishing sieve

The pulse jet was intended for flying speeds of at least 700 km/h. We therefore thought that we could supply the air at the corresponding dynamic pressure. We told ourselves: If one of the two components needed for the explosion, i. e. the air, enters intermittently, the other component, i. e. the fuel, can be supplied continuously to the combustion chamber. This analysis facilitated the development of our pulse jet and its control system considerably, and later on contributed to the simplification of the power plant of the flying bomb.

\* Dr.-Ing. — Formerly: Director, Argus Motoren Gesellschaft, Berlin, Member of the V-1 Working Staff. — At present: Director, Ernst Heinkel Fahrzeugbau AG., Stuttgart-Zuffenhausen, Germany.

When pulsation took place, two things had to be prevented:

- (i) A return flow of the combustion gases. We had therefore provided a flow-technical valve in the form of a BORDA mouth.
- (ii) A continuous burning of the fuel. For this purpose the atomizing nozzle was sunk into a small secondary chamber, and this chamber was screened from the combustion chamber by a flame extinction strainer (based on the principle of the well-known miners' lamp).

The apparatus (Fig. 2) was first operated on November 13th, 1939, and immediately we were surprised to observe an intermittent operation with pulsations of high frequency.

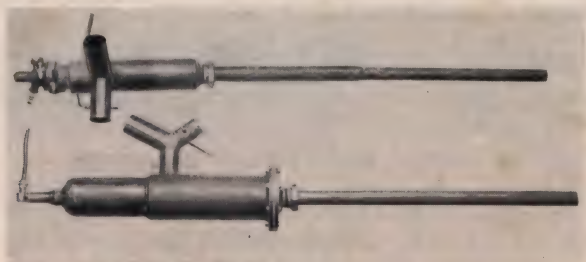


Fig. 2. Models corresponding to Fig. 1

On a second model (Fig. 3), the air entered from the front, under the ram-pressure, and was deflected into the combustion chamber via an annular vortex. We also aimed at achieving an annular vortex in this ball-shaped combustion chamber. The combustion of this model was excellent, and its pulsating operation was steady. We were surprised, however, by the fact that the apparatus continued working satisfactorily after we had switched off the ignition.

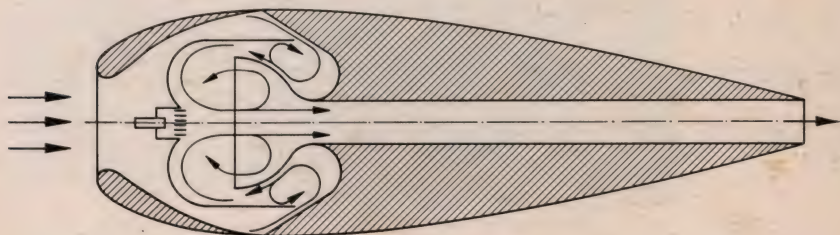


Fig. 3. Second model of the pulse jet with air intake at front

Our third model was equipped with a leaf-spring valve which could be bought as a standard product, i. e. a compressor valve. This pulse jet (Fig. 4) was designed as follows: Fuel and air enter the mixing chamber together, without deflection and in the same direction. The effect of the flame extinction strainer was now achieved by means of a necked portion. The velocity of the mixture became so high, and the pressure in the secondary chamber was balanced in such a way, that a flash back of the flame and a continuous burning

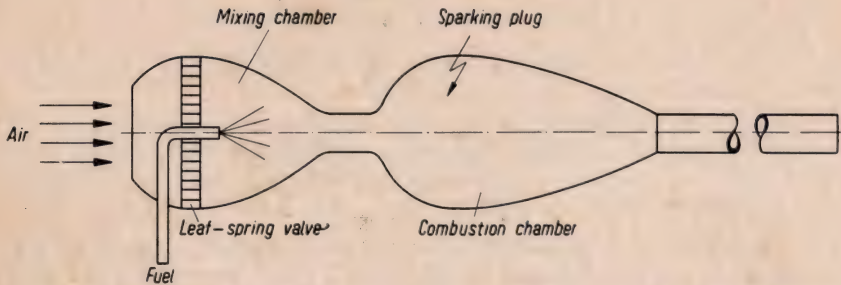


Fig. 4. Third Argus pulse jet model. The neck-down in the combustion chamber prevents continuous burning of the fuel at the injection nozzle

of the mixture were prevented under all circumstances. Moreover, this device protected the sensitive valve from being directly touched by the burning gases.

Hence this apparatus already contained all the elements of our nozzle-diaphragm — mixture-formation — process, which will be dealt with later on.

After reaching this state of our experiments, three and a half months after starting them, we were informed by the Ministry that the pulse jet had already been the subject of investigations of another group in Germany for several years. It was the first time that we heard the name of PAUL SCHMIDT, and we were asked to inspect his work at Munich. Herr SCHMIDT demonstrated his big pulse jet (Fig. 5) in March 1940. A big paper bag filled with air and propane gas was suspended in front of the valve apparatus and the pulse-jet unit operated some seconds until the gas was consumed.

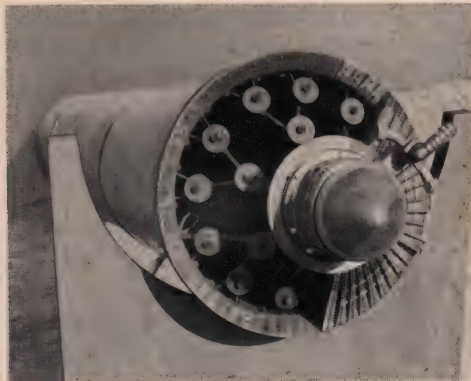


Fig. 5. Conical valve apparatus (see also Fig. 2) of the pulse jet SR 500 built by Paul Schmidt. Destroyed after 13 minutes running time with 450 kg thrust

Herr SCHMIDT had planned to use a small pulse jet for the operation with liquid fuel, but the liquid-fuel system was then obviously in its early state of development.

## 2. LOW-SPEED FLIGHT TESTS

Up to that time Herr SCHMIDT had only achieved short-time operation on test stands. Nobody knew what thrust could be achieved by means of these

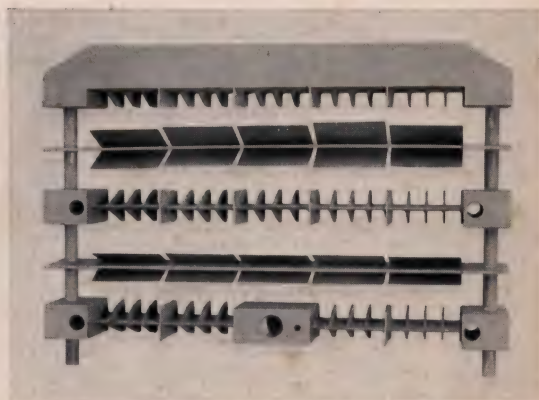
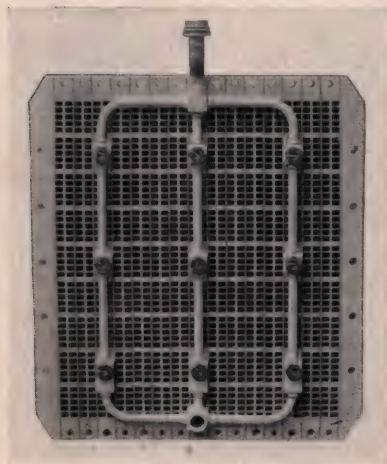
pulse jets during flight, or if the further development for the propulsion of high-speed aeroplanes was worth while.

Therefore, ARGUS was asked to develop a pulse jet with a static thrust of some 120 kg for investigations during flight. This was to be done in the shortest possible time, and on April 30th, 1941, this pulse jet made its first flight suspended beneath a training plane (Fig. 6).



Fig. 6. 30th April 1941: First flight of a pulse jet (Argus design)

The whole valve system, however, was arranged level (Fig. 7), for simplified manufacture, and we adopted from Herr SCHMIDT the element of the preliminarily bent valve spring flap (Fig. 8). As to the mixture formation we adhered to the method developed by ourselves.



Figs. 7 and 8. The flat valve apparatus of the Argus pulse jet and constructive arrangement with fuel atomizer nozzles and starting air duct

In the summer of 1941 we were asked to motorise cargo gliders. In this connection we arrived at the first aeroplane driven solely by pulse jets.

Towards the end of 1941 the pulse jet had proved to be satisfactory for low-speed aeroplanes, but it was not at all clear if the unit was suitable for higher flying speeds. Doubts were raised in this connection which had to be taken seriously.

At that time my collaborator, Dr. DIETRICH, who had earned great merit in initiating the development, separated from ARGUS, after he had proved, as he thought, in a memorandum, that no useful thrust could be expected from jets at speeds exceeding 600 km/h \*.

### 3. THE V-1 IS ORDERED

Aerial warfare became more and more difficult, and the argument of insufficient accuracy, which had been the reason for the refusal of the long-distance flying bomb at the beginning of the war, became invalid. We therefore decided to submit the project once more and to suggest the pulse jet as its engine.

In view of the fact that, at that time, the performance of the pulse jet at high speeds was completely unknown, it was a bold decision of the German Air Ministry to order a long-distance missile on June 19th, 1942.

The decision to equip an unmanned flying bomb with a pulse-jet engine raised additional problems for ARGUS. These were:

- a) Determination of the thrust of this unit at high flying speeds;
- b) Design and construction of a fully automatic fuel-control system;
- c) Development of a suitable starting system in order to allow the launching of the projectile by pressing a button.

### 4. POWER PLANT OF THE V-1

While ARGUS worked at these tasks, considerable progress was achieved with the design of the airframe. The lay-out of the power plant is shown in Fig. 9.

\* In order to set right numerous misunderstandings found in technical literature, I herewith wish to clarify the following points:

1. The pulse jets of Messrs. Argus were not constructed under licence but were an independent development.
2. Argus pulse jets, i. e. including "V-1" propulsion, did not employ the Schmidt patent 523 655 (of 25th April 1931) which was, erroneously, frequently mentioned in this connection.
3. This patent 523 655 does not at all deal with the original idea of the invention of the pulse jet itself, which had been known since 1906, but concerns an additional idea in saying that in "known, approximately pipe-shaped reaction spaces, the one end of which is open" (Karavodine, Marconnet), "a quantity of air, the weight of which is from 10 to 50 times greater than the weight of the inflammable mixture, is immediately accelerated by the force of the excessive pressure of the exploding mixture...".
4. The realization of this idea was successful only in individual explosions, however, never, in repeated continuous operation. After years of fruitless test work the inventor (Schmidt) gave up his efforts to achieve continuous explosion, as early as 1938.
5. Laboratory pulse jets, as Herr Schmidt had since operated and demonstrated in March 1940, function in principle in accordance with the ideas, patents, tests and publications by Karavodine, Marconnet and Barbezat (French patents 374 124 and 412 478) which had been known since the beginning of this century. (Literature: "Die Turbine", publishing year 1909, page 305 and subsequent pages, and "Stahl und Eisen", published 1911, No. 42, page 115 and subsequent pages.)
6. Marconnet had even laid down the plate spring valve in his patent specification (French No. 412 478) as the inlet device for the pulse jets: "The mixture enters the combustion chamber through a very light valve... This valve consists of a metal plate spring which opens and closes similar to the voice of a clarinet...".
7. However, whilst Marconnet's plate valve was flat when in a closed state, and curved when opened, Herr Schmidt had recommended the reverse. His valves were curved when in a closed state and flat when the valve was open. This "precurved" plate valve was the only device which Argus really took over from Schmidt, whilst the valve apparatus as a whole was completely and basically remodelled by Argus, and with progressing development was more and more simplified to facilitate production.

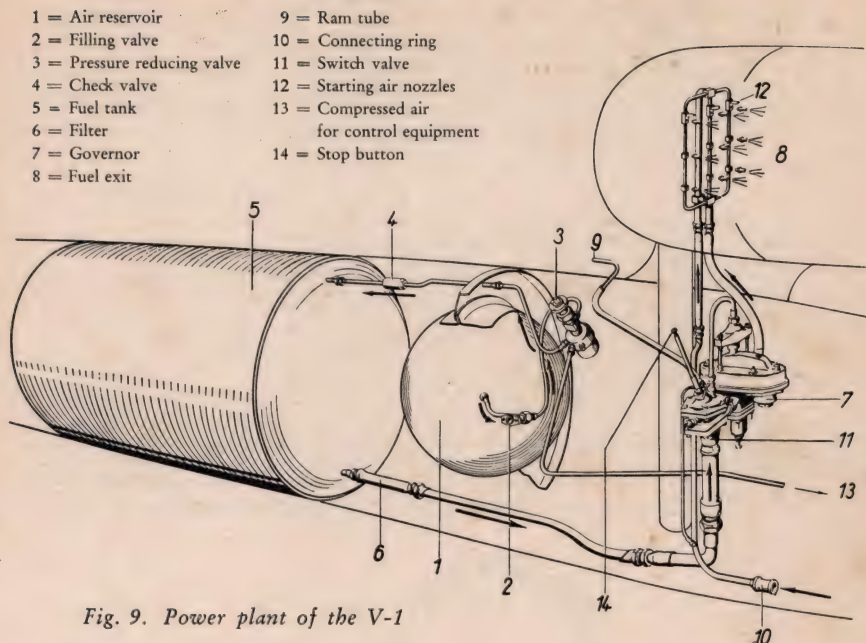


Fig. 9. Power plant of the V-1

#### 4.1. Fuel Supply

The fuel in the tank was put under pressure by means of the compressed air which in any case was needed for the control system. The fuel flowed through the filter and control system, was atomized by the injection nozzles, and was continuously supplied to the combustion chamber.

#### 4.2. The Fuel-Control System

With this unmanned flying body the fuel-control system had to perform complicated tasks, which can be judged from a consideration of the time histories of the fuel tank, and a point just ahead of the injector nozzles (Fig. 10).

Before the take-off the pressure in the fuel tank was 7 atm but for starting the injection pressure at the nozzles had to be reduced to 1.2 atm. Immediately after starting, the injection pressure had to rise to 2.2 atm for static operation, while the pressure in the tank simultaneously decreased to 6.8 atm.

Owing to the inertia effect of the fuel columns, a pressure peak of 9 atm arises when launching, and this pressure peak has to be removed by the control system.

During the launching operation the pressure at the nozzles had to rise simultaneously to 2.6 atm, corresponding to the take-off speed.

The injection pressure then had to drop during climb and rise again with increasing flying speed after changing to horizontal flight. During flight the pressure acting on the fuel tank slowly dropped from 7 to 6 atm.

Fig. 11 shows the design of the fuel-control system. When the stop valve was opened, the fuel flowed to the constant-pressure valve which had to maintain a constant pressure of 4 atm in front of the throttle valve. The throttle valve

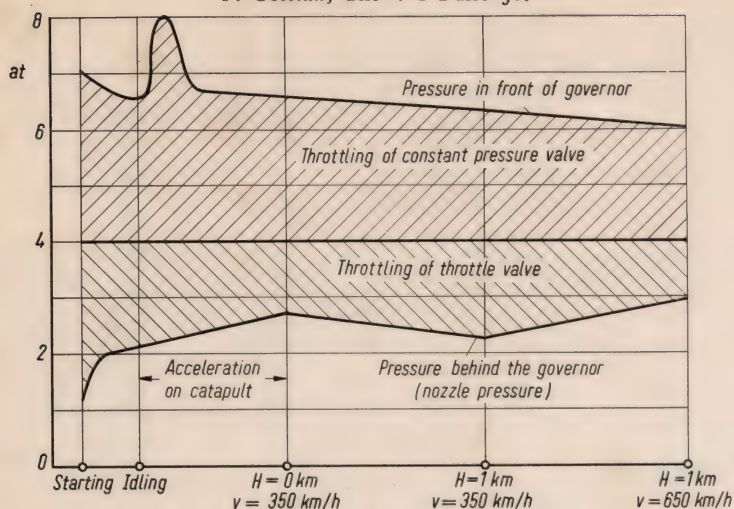


Fig. 10. Task to be accomplished by the engine fuel-control system

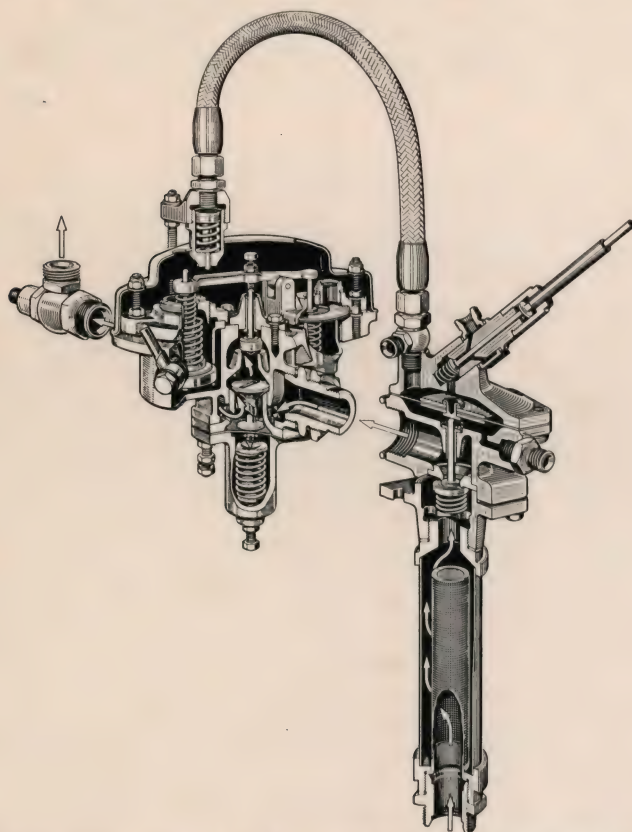


Fig. 11. Fuel-control system of the V-1. Filter with locking valve and stop mechanism, balanced-pressure valve and throttle valve, ram-pressure piston and barometer capsule

was subject to the influence of the ram-pressure on the one hand and of the altitude capsule on the other hand, both of them acting against each other on a balance beam. The altitude capsule was a chamber which was closed by a plastic diaphragm. With increasing altitude, the fuel flow was throttled and with increasing ram-pressure the throttle valve was opened wider.

#### 4.3. The Push-Button Starter

The starter system of the power plant was connected with the fuel-control system. We had found out in preliminary tests that, for safe starting of the pulse jet, it was important to bring a certain fuel quantity with a corresponding quantity of compressed air simultaneously and suddenly into the combustion chamber, so that no burning, but rather a violent explosion, should take place.

When compressed air was let onto the diaphragm of the stop valve for starting, it opened suddenly. At the same time the compressed air on the balance beam of the fuel-control system acted in such a way that the injection pressure was reduced to 1.2 atm. This starting method was very reliable and enabled us to use the push-button starter.

### 5. FLIGHT-TESTING OF THE V-1

#### 5.1. The First Test Shot

Work on completing the first test V-1 had made extraordinarily quick progress, and the first test shot was fired on December 24th, 1942 — six months after the initial order. At this time no results of the flight measurements on the pulse jet, which would have rendered possible an optimum design of the governing system, were available.

During the flight tests the pulse jet proved to be rather a troublesome power unit. The sensitive carriers could not stand the high pressure fluctuations of the pulsating jet, and long repair work on the frames had to be carried out after each test flight\*.

When the fuel-control system had to be supplied in the winter of 1942 we only knew that the pulse jet extinguished easily with too high a rate of fuel flow. In order not to endanger the initial tests of the V-1 the rate of fuel flow was controlled with the greatest precaution, and the flying speed was not at all satisfactory in the first shots. But progress was achieved, and speeds of some 600 km/h were soon reached.

#### 5.2. Speed Crisis

It was just at the time of the invasion when a serious reaction set in. Peenemünde announced that the flying speed of the test equipment suddenly

\* Practical flight tests carried out by the Argus Flight Department were of decisive importance for the airworthiness development of jet propulsion, and I wish to express in this place my special thanks to our Flugbaumeister Staeger and Schenk for their valuable co-operation. In the evaluation of the results, Dr. Zammert and Dr. Flössel, and in the constructive development of the various fuel-control systems which were based on these evaluations, Messrs. Weiche, Belitz and Kreuziger earned special merit.

dropped to 450 km/h. As you can imagine, we were thoroughly upset. We were thinking of the cause day and night and finally found out: the altitude capsule of our fuel-control system had been covered by a plastic diaphragm. Without our being informed, this plastic material had been changed to another material which allowed the fuel to diffuse. Gasoline sometimes flowed over this diaphragm when testing the governing system, and this gradually diffused into the capsule. The altitude cell thus worked incorrectly and hence too much fuel and power were throttled at low altitudes.

A great number of low-speed V-1's were actually launched.

### 5.3. Increase of the Speed from 600 to 765 km/h.

We, the engineers, did of course not agree with this decision and we were finally permitted to substitute new altitude capsules. The speed was in this way increased to 645 km/h.

The results of our flight department had meanwhile shown that the rate of fuel flow would allow another increase of speed.

That was why, in August 1944, the ram piston in the fuel-control system was enlarged. The speed of the V-1 increased by another 50 km/h, but still higher rates of fuel flow led us to expect further improvements. We now modified the constant-pressure valve so that a higher injection pressure became available at the fuel nozzles. The result of this measure was another 75 km/h increase of the flying speed.

In order to achieve an optimum (Fig. 12), a new very simple fuel-control system was designed in which the constant-pressure valve was omitted and the

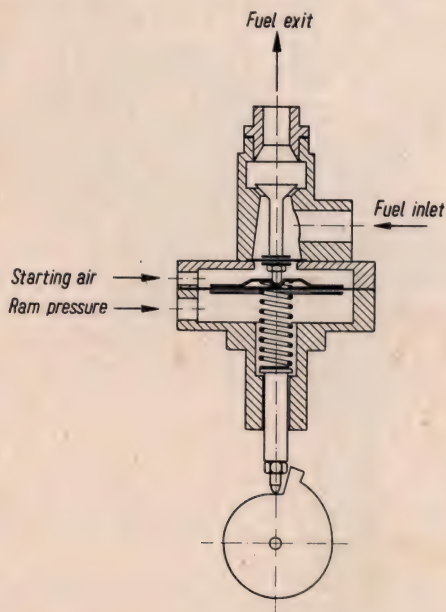


Fig. 12. Essentially simplified fuel-control system (end of 1944); adjustment of flight attitude by a cam

work of this valve was carried out by the throttle valve. Owing to the omission of the corresponding throttle losses we succeeded in using the full pressure of the fuel in the tank at the nozzles. Another 25 km/h of speed increase was thus obtained.

Fig. 13 shows the results. The upper curve shows the limit of the highest thrust obtained without superfattening. This limit was found by flight-testing.

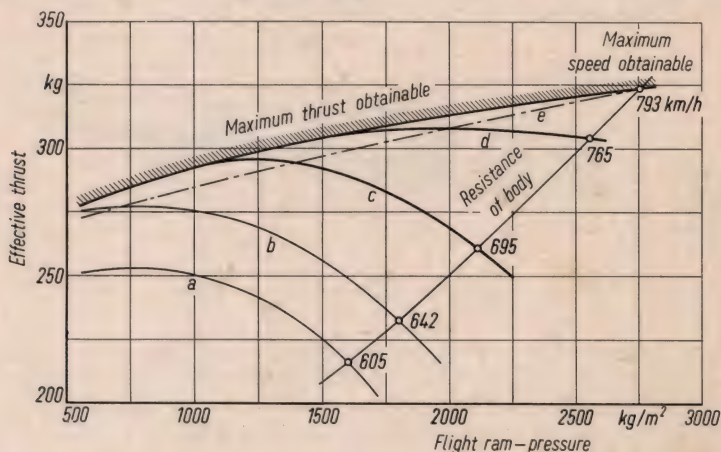


Fig. 13. Increase in flying speed of the V-1 from 650 to almost 800 km/h by adjusting the fuel-control characteristics to the possible fuel flow of the pulse jet As-014 which had been determined in flight tests

Other curves show the drag of the airframe and the thrusts achieved by means of the various types of fuel-control systems.

Only a few test types of the fuel-control system corresponding to curve *d* were constructed. The flying speed reached with them was 765 km/h.

We constructed only a few specimens of the single-valve fuel-control system (curve *e*). The results of measurement were reported in the last session of the V-1 Working Staff on February 2nd, 1945. The speed of the V-1 had been increased to 800 km/h!

However, the war came to an end before these improvements had reached the front. They had been achieved by improving the fuel-control system only, but the pulse jet itself had not been modified.

## 6. THE LAST TYPE OF THE ARGUS PULSE JET (1945)

### 6.1. Design

Fig. 14 shows that type of the pulse jet which was finally used for the V-1. The 3 m long pipe with the combustion chamber and the spark plug, the jet diaphragm, the valve box, the fuel nozzles with the fuel line and the intake cowling with the fork suspension, can be seen at bottom left.

The upper part of the photograph shows a sectional view of the head of the pulse jet. The line of the compressed starting air and the small tubes supplying

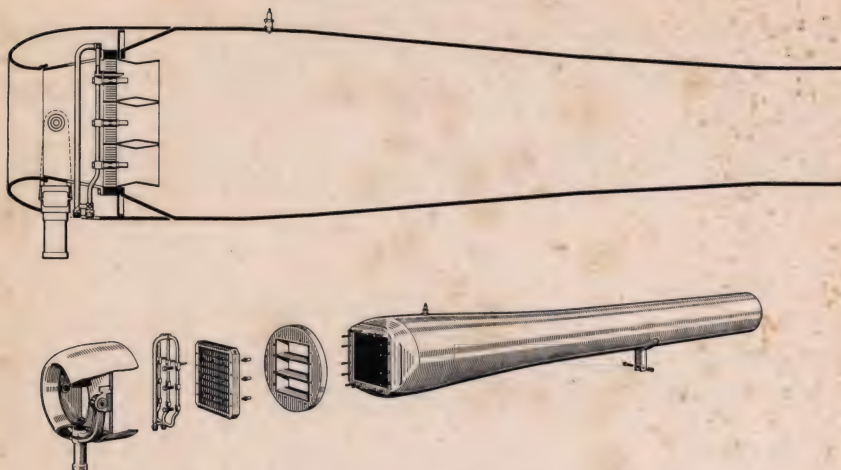


Fig. 14. The Argus pulse jet as constructed for the V-1 (1944/45).

the starting air under the upper three fuel nozzles can be seen. The fuel nozzles were constructed as swirl atomizers. At maximum speed they worked at a pressure of some 3 atm; good atomization was, however, also achieved at only 1 atm. These swirl atomizing nozzles contributed considerably to the good controllability and to the reliable starting of the pulse jet.

It was proved that the pulse jet could be controlled in such a way that it gave only 10% of its full thrust without any irregular operation occurring.

The mixture formation plant was similar to that of our first model pulse jets. The fuel nozzle was also housed in a secondary chamber. The flame extinction was achieved by the increased mixture speed at the necked-down portion. It was proved that the nozzle diaphragm protected the sensitive valve spring flaps and the die-cast light-alloy nozzle webs from thermal over stresses, as the flame obviously did not reach the valve system.

## 6.2. Dimensions and Weights

This type of pulse jet gave a static thrust at sea level of 350 kg. Its weight was 138 kg for a length of 3.6 m. We achieved a thrust on the ground of 350 kg for a consumption of 0.8 g/kg.

## 6.3. Thrust and Consumption Curves

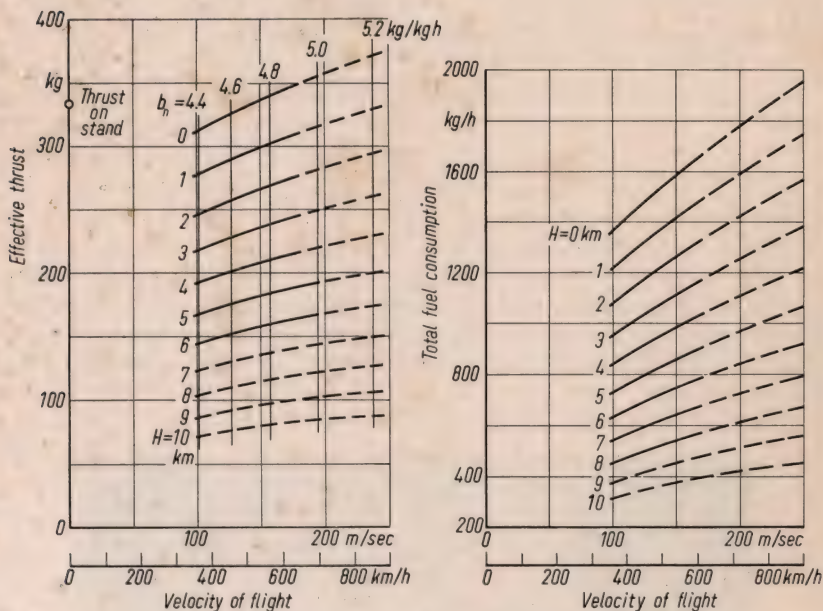
In June 1944 our flight-testing department had completed all the measurements which enabled us to draw the thrust and consumption curves.

Fig. 15 shows the thrust as a function of altitude and speed. The thrust continued rising with the speed in the measured range. Below 350 km/h the thrust was less than the static thrust at sea-level. We found later on that a diffuser at the end of the pipe was useful in this lower speed range, while it was disadvantageous above 400 km/h.

The absolute fuel supply increased according to the higher rate of air flow with rising speed (Fig. 16).

Our measurements had proved that the specific fuel consumption depended only on the flying speed (Fig. 17).

These were our last systematic flight measurements.



Figs. 15 and 16. Thrust and consumption of the pulse jet As-014

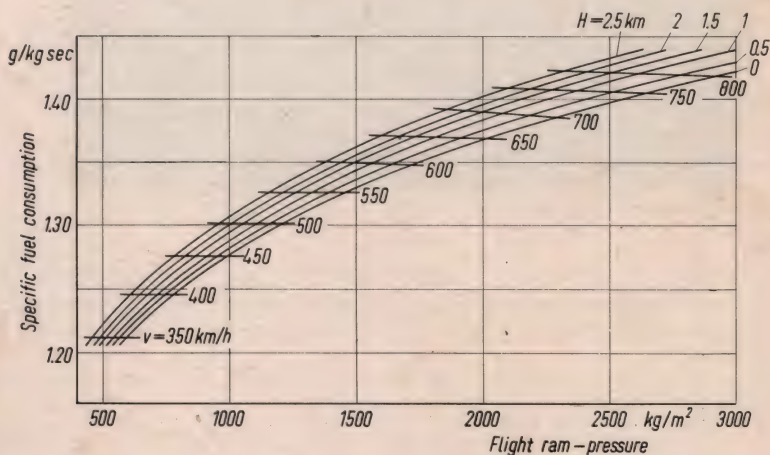


Fig. 17. Specific fuel consumption of the pulse jet As-014 at maximum thrust

## 7. THE FUNDAMENTAL WORKING PROCESS OF THE PULSE JETS

### 7.1. Ignition

The pulse jet was known to continue working after switching off the ignition. The phenomenon of self-ignition had already been observed by BARBEZAT. The following remark on this subject was published in the periodical "Die Turbine" in 1909:

"The new charge spontaneously enters the explosion chamber without any control and this process repeats automatically within a very short time. The upper part of the chamber and the nozzle pipe soon began to glow and the ignition could be stopped . . ."

This reference was interpreted in such a way that one could think the gas had ignited at the red-glowing wall.

A portion of our V-1 pulse jet began to glow, too, and we therefore had to ask whether the ignition were also started by the glowing wall in our case. Water-cooling was provided for the endurance tests of our pulse jet on the ground (Fig. 18). Under these conditions, however, the wall certainly did not glow.



Fig. 18. Endurance test of the Argus pulse jet on the ground with water cooling

The periodically switched-off or switched-on water-cooling showed that there was no difference in operation of the jet pulse with a glowing or with a cold wall. We therefore assumed that the wall temperature did not influence the ignition.

### 7.2. Gas-Exchange Process

It remained to be investigated how this simple apparatus could periodically suck in a new charge without needing any mechanical devices (as e. g. pistons, etc.). This subject had also been discussed in technical journals as early as 1909.

Then it was asked whether or not the cooling down of the combustion chamber played a part.

It was explained later on that the burning gases left the pipe like a piston and sucked in the fresh charge behind them. Still later on it was clear that the pulse jet worked in resonance. But this is a summarising explanation only.

We made it our task in the ARGUS Company to find out in detail the physical processes inside the pulse jet and we first succeeded in doing so in May 1941. We calculated the processes and showed them in an animated film \*.

## 8. APPRECIATION OF THE PIONEER WORK

If we estimate the whole importance of these processes for the principle of the plant, we shall find that the long pipe is the essential part of the pulse-jet unit. The physical processes which are necessary for the working principle take place in this pipe and the ingenious thought of such a pulse jet is obvious. A very long pipe which is shut off by a valve on one side and which is acoustically excited by pulsations can suck in and discharge gases without the aid of any movable parts and can thus carry out a thermal cycle without any ignition device.

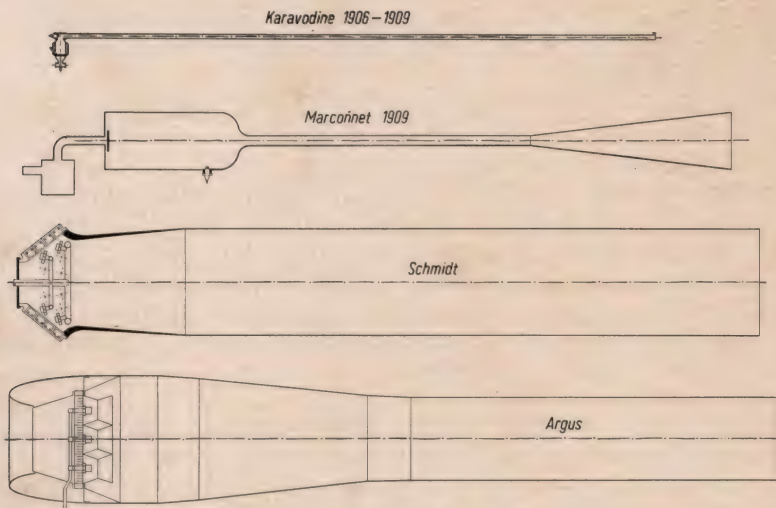


Fig. 19. The historical development of the pulse jet from 1909 to 1945

The realization of flying speeds of more than 750 km/h by means of so simple an apparatus is the success of an idea, and I think that we should remember the inventors and the pioneers of this peculiar engine, who are (Fig. 19):

\* This strip demonstrates the movement of the waves in the pipe; it shows in particular how the pressure wave arriving from the combustion chamber turns at the open end of the jet, moves back as an expansion wave, opens the valves, and sucks in the fresh air load.

KARAVODINE, who talked about his idea in 1906 and who made endurance tests of many hours with a turbine;

BARBEZAT, who discovered and published in 1909 the principle of self-ignition;

MARCONNET, who suggested the use of the pulse jet as an aero-engine in 1909 and who was thus far ahead of his time;

PAUL SCHMIDT, whose merit and struggle for this unit which had gone on for many years will not be forgotten in the history of engineering.

The ARGUS Company was given the opportunity to develop this unique engine until it reached the stage where it could be used in a flying body. Thus ARGUS contributed to the realization of the long-distance missile in aeroplane form which had been suggested by the author in the first days of World War II.

## DISCUSSION

Dipl.-Ing. PAUL SCHMIDT (Munich): The "tedious repairs to the bodies", which Dr. GOSSLAU quoted in his lecture as having been necessary after each test flight, were probably originated by insufficient elasticity between engine and body. This was, at any rate, the case with the V-1.

The Supreme Command of the LUFTWAFFE informed me in January 1945 of tests which were run during 1944 and which had shown strong oscillations of the bodies. It stated: "Moreover, it showed that the amplitudes decreased with increasing temperature of the pulse jet, but by then the navigation system had already been destroyed."

Several months before, I had received three or four pulse jets with their supports for testing purposes and I had found that the stiffness of the connection system between pulse jet and body was far too hard, and this resulted in a dangerous oscillation condition. Impulses with the period of the pulse jet could act on the body with a force of several thousand kilograms.

In 1942/43 the REICHSLUFTFAHRTMINISTERIUM had ordered an investigation on the previous works of MARCONNET and KARAVODINE quoted in the lecture, to find out if the power system of the V-1 was to be considered as a new type of engine. The results of this investigation led the ministry to call the engine of the V-1 the "ARGUS-SCHMIDTROHR".

Dr.-Ing. F. GOSSLAU: Based upon my remarks regarding the tedious repairs to the bodies, Herr SCHMIDT claims that the propulsion unit of the V-1 was not sprung softly enough against the body. In this connection he refers to an information obtained from the Supreme Command of the LUFTWAFFE in January 1945 concerning relevant tests in 1944.

I am under the impression that there errors in fact and in time are apparent.

Referring to the memoranda and notices which are still today in my possession, I want to reply as follows:

The propulsion unit was sprung against the body by voluminous rubber bumpers, the stiffness of which was altered in joint tests of the firms of ASKANIA and ARGUS so long until detrimental effects upon the control system were eliminated. These tests were performed on the test-stands of ARGUS at

Berlin from 22nd till 29th May 1943 for the compass. The main task of the "Working Committee V-1", which was strictly guided by the Air Ministry, was the permanent supervision of the operational safety and the permanent improvement of new weapons. If a fundamental constructional fault regarding the elasticity of the propulsion unit would have existed, quick and severe action would have been taken and an alteration would have been performed.

In fact, however, the damages to the bodies mentioned in my paper did not refer at all to the V-1, but to the manned test-carriers of our flight-test department (i. e. DORNIER and JUNKERS aircraft), and besides the damages occurring there had nothing to do with the transmission of thrust.

With those damages as mentioned by me we met for the first time in connection with transport gliders. Here the covering of the body consisting of woven material was often torn away in rags by the considerable variations of pressure caused by the rhythmic exhaust-bang of the operating pulse jet mainly near its outlet. Thus it was not the variation of thrust, but the acoustics (due to the working-principle of the pulse jet) the true cause of the damages.

Like the transport gliders, also the metal sheeting of the bodies of our test carriers (Do 17, Ju 88) was affected, and these repairs delayed so frequently the continuation of the flight tests.

The often misunderstood and criticized arrangement of the propulsion unit of the V-1 at the end of the body above the side control, and the duct which protruded far beyond of the end of the body were important factors for avoiding such damages with the V-1.

In brief, I would like to say to this subject that it was not the transmission of thrust which took so much of our time, but the enormous acoustics of this unit which damaged not only the lightly built aircraft, but even the brickwork of wind tunnels.

Prof. Dr. H. BLENK (Brunswick): I would like to add a couple of words to Dr. GOSSLAU's lecture. It must have been in 1942, when the LUFTFAHRTFORSCHUNGSANSTALT in Braunschweig-Völkenrode got an urgent inquiry from Peenemünde as to whether the ARGUS pulse jet could be tested in one of our wind tunnels. What had happened in Peenemünde was that the firing tests at high velocity showed that there was no longer a propulsive force. Obviously thrust and resistance compensated one another, so that the resultant force became zero. In Braunschweig we had a high-speed wind tunnel going up to MACH number 0.9, with a test diameter of 2.80 m, so that a full-size test pulse jet could be installed. Tests at those speeds actually revealed that the propulsion was nil, which means that the thrust equalled the resistance. In collaboration with the firm of ARGUS we then altered the pulse jet, especially the intake, from day to day and performed new tests. The firm of ARGUS based the alterations mainly on the proposals of Dr. ZOBEL — as far as I remember — with such haste that alterations were done within the day, and the new tests were done the same night. The wind tunnel tests led to a form which provided the necessary thrust for the V-1 at speeds of up to 700 km/h.

Prof. Dr. ERNST SCHMIDT (Munich): In his remarks dealing with the difference between the measurements of the effective thrust of the pulse jet placed on an aircraft and in the Völkenrode high-speed wind tunnel, Prof.

BLENK did not tell the whole story. I remember it in more detail. A careful investigation of the discrepancies of the measurements showed that the aircraft measurements which Dr. GOSSLAU mentioned in his report were actually wrong for the following reasons: the pulse jet was fixed above the carrier aircraft with the help of a frame and the thrust was measured electrically and integrated at the same time.

However, the integrating device worked quadratically and in this way the negative values of the thrust which really existed during part of each period of the pulsations were converted into positive ones. Thus, in contrast to the measurements, the effective thrust at the intended velocity of flight went down almost to zero and the whole V-1 would have been a complete failure if TH. ZOBEL had not succeeded in aerodynamically improving the air intake by adding an entrance diffuser with a well rounded mouth. Only this favourable chance made it possible for the design flight speed to be reached.

Dipl.-Ing. PAUL SCHMIDT: The remark made by Prof. Dr. ERNST SCHMIDT regarding an error in the measurement of thrust due to the periodical occurrence of a negative thrust points to too hard a springing between tube and body. It emerges from the curve in Fig. 14 of my paper (see page 387) that in the case of a too hard springing there will occur with the periodically generated positive thrust of 2000 kg a periodically occurring negative thrust of 1500 kg. However, with adequately soft springing no negative thrust is observed.

Dr. GOSSLAU: I would like to combine my replies to the remarks brought forward by Prof. BLENK and by Prof. E. SCHMIDT in this discussion. As I have mentioned before, work on the V-1 was speeded up with utmost energy and by all means available after the order was placed on June 19, 1942. In this connection the LUFTFAHRTFORSCHUNGSANSTALT in Braunschweig was asked in October 1942 to test the ARGUS pulse jet in the high-speed wind tunnel. A fast test plane equipped with an electronic device for thrust measurements was assigned to the ARGUS flight test department.

From this point there began a series of tragic errors which almost resulted in failure of the whole V-1 project. Braunschweig reported a disastrous decrease of thrust down to zero at 600 km/h. The thrust measurements of the ARGUS flight test department were wholly incomprehensible. The more we tried to reduce the resistance of the pulse jet, the worse became the effective thrust.

In this rather dangerous situation the German Air Ministry summoned, in the autumn of 1942, a committee of some 20 experts on research and production, the "Working Committee on Jet Propulsion Units". Here the results obtained by the LUFTFAHRTFORSCHUNGSANSTALT in Braunschweig and by the flight test department of ARGUS were discussed. The most important problem was: why does the pulse jet stop operating at high speeds of flight (in the Braunschweig wind tunnel)? — The general opinion was that at high speeds the flame was blown out of the pulse jet at its rear end which would terminate the working process. This statement was contradicted by our flying tests; yet the thrust measurements during the flying tests remained inexplicable. A pulse jet which was fitted with numerous collars on its outside to increase its rigidity, and which thus should have shown a high resistance, showed better effective thrusts than the same pulse jet without such collars. In view of this really senseless result, towards the end of November 1942 I sent Dr. VOLLAND, our expert on

measurements, to the flight test department with the order to check critically the measuring equipment. He found out on the same day that the measuring device recorded the negative thrusts, which necessarily occur in any working cycle, erroneously as positive values. This information was passed on by me on December 1, 1942 to the Working Committee on Jet Propulsion Units to which, as far as I remember, Prof. E. SCHMIDT also belonged, and his here remark probably refers to this fact. The measuring error was the fault of ARGUS, it was, however, also discovered by ARGUS.

Following the adjustment of the measuring device, the discrepancies in the results obtained in the Braunschweig wind tunnel became larger than ever before. Braunschweig insisted that the thrust would decrease to zero at 620 km/h; flight tests carried out by ARGUS revealed an effective thrust of about 300 kg at this speed. This was the state of affairs when, on December 24, 1942, the first test shot was fired which, apart from the unsatisfactory speed, was successful.

ARGUS then decided to use its own so-called blower test stand, i. e. a sort of wind tunnel generally used for testing air-cooled engines, for testing the pulse jet. A surprising result was obtained on the very first day of test. The relevant entry in the diary read as follows: "With a blowing velocity of 300 km/h the first test runs showed a thrust of 170 kg only. Running without blowing resulted in a thrust of 200 kg only. On the open-air test stand the same pulse jet showed the normal thrust of 320 kg."

The weekly report for the period January 6 to 13, 1943 stated: "The pulse jet measurements on the blower test stand were stopped due to the lack of coincidence with the measurements in the open, and following heavy damage caused by the tests to the brickwork."

In Peenemünde a wooden hut had been erected where the V-1 performed short test runs prior to being launched. After only a few such test runs there nothing but the timber frame existed. The walling had fallen off completely due to the effects of the operating duct.

I hope have shown by these remarks that in closed rooms a pulse jet can not be correctly tested as to its thrust. Any wind-tunnel measurement is bound to lead to faulty results, and this applies also to the measurements in Braunschweig quoted by Prof. Dr. BLENK. Of course, these critical remarks refer only to the pulse jet in operation. The cold resistance may be determined satisfactorily in a wind tunnel.

When, in May 1943, various intake-diffusers were suggested by Prof. Dr. BETZ, Dr. ZOBEL, Prof. Dr. RUDEN and by ARGUS, the diffuser proposed by Prof. Dr. RUDEN showed the best results with a gain of thrust of about 60 kg at 650 km/h. There can be no doubt that the effective thrust of the ARGUS pulse jet in the V-1 amounted from the very beginning to about 215 kg at 600 km/h (Fig. 13). The further increase of the effective thrust, resulting in the speed of the V-1 increasing to 790 km/h, has to be attributed to the knowledge that the pulse jet would stand a much higher rate of fuel flow than could be assumed in view of the initial lay-out of the governor. We owe this perception to the flight test department of ARGUS, to the Flugbaumeister STAEGE and SCHENK, to the evaluation of the results obtained by their flights by my collaborators Dr. ZAMMERT and Dr. FLÖSSEL, and finally to the constructional and experimental work on the adaption of the fuel governor for which Messrs. WEICHE, BELITZ, KREUZIGER and GEILER were responsible.

Admiral FAHRNEY (Philadelphia): The V-1 missile resembles, in most of its characteristics, the "Flying Bomb" project in America in World War I and the "Assault Drone" project in World War II. It was planned to use radio control in the earlier project and it was used successfully in the latter project. Since you were able to get good triangulation stations along the coast, why did you not use radio control in the V-1 for greater accuracy of directed flight?

Dr. GOSSLAU: Owing to tragic impressions received at the front towards the end of the first World War, the idea of an unmanned, teleguided aircraft has haunted me ever since. In April 1937, in agreement with the ARGUS factory, I submitted to the DEUTSCHE FORSCHUNGSANSTALT FÜR SEGELFLUG (DFS) and to the C. LORENZ AG, a scheme for collaboration on the subject of radio-controlled flying bodies which were not to carry persons, but only technical equipment.

The teleguided anti-aircraft target model, whose teleguiding equipment has been described by Dr. KLOEPFER was the first outcome. This model performed quite a successful teleguided flight on May 14, 1939 and may be regarded as the first teleguided flying body in Germany for military use between the two World Wars.

When, on November 9, 1939, I submitted to the Reich Air Ministry my first memorandum regarding a power-driven winged bomb having a range of about 600 km, I had of course visualized radio control. But it was just this proposal which delayed for so long the realization of the flying bomb. Electronics was one of the decisive bottlenecks of that period. Furthermore, interference and possible tracking by the enemy were feared.

Only when the Channel-coast was under German occupation and when compass guidance alone promised satisfactory hitting accuracy on large target areas, was the realization of this idea decided upon. Later on, as the V-1 actually flew, the idea of its teleguiding was taken up again, as Prof. FISCHER has told this assembly.

## RESEARCH AND DEVELOPMENT OF THE GUIDED MISSILE "FEUERLILIE"†

WALTER WERNITZ \*

The idea of the "FEUERLILIE" research programme was born in 1941—1942. Its object was the determination of aerodynamic and stability data at high speeds, especially in the vicinity of the speed of sound. These results were very important for the development of high speed aeroplanes and guided missiles, and were also to be used for comparison with high speed wind tunnel tests. Until the end of the war the "FEUERLILIE" programme was lead by the Research Department of the German Air Ministry.

At first it was intended to use the projectiles made by the aircraft industry (i. e. Hs 293 etc.) for these tests, but there were two facts which made this impossible. No additional specimens could be made available for this research programme and the outlines or shapes of these projectiles did not satisfy the requirements of high speed research.

Therefore Dr. GERHARD BRAUN and Prof. BUSEMANN, LUFTFAHRTFORSCHUNGS-ANSTALT (LFA) at Braunschweig-Völkenrode, produced a design with a swept wing and a double and symmetrical elevator control. All the different sizes of the "FEUERLILIE" had approximately the same aerodynamic shape (Fig. 1).

The scope of the programme was stationary flight controlled by an automatic pilot (development by Dr. KERRIS, LFA) and differential elevator steered motions, in which the automatic pilot was to keep the missile in a vertical plane. For gliding and differentially steered flights, the "FEUERLILIE" was started from and aeroplane of the test-station of the German Air Force (VERSUCHSSTELLE DER LUFTWAFFE) at Peenemünde-West.

In the first tests a 5 cm diameter "FEUERLILIE" was launched, from the ground, with the aid of small powder rockets. In the second stages (F 25) the diameter was increased to 25 cm and launchings were made at Leba (Pommern) along a steep slideway, using larger RHEINMETALL-BORSIG powder-rockets.

In the same manner, but on a larger slide-way, later launchings were carried out on the Greifswalder Oie island near Peenemünde with a 55 cm diameter "FEUERLILIE" (F 55) driven by a liquid fuel rocket developed by LUTZ and NOEGGERATH. The F 55 was also tested in a tailless form.

The flight of the F 25 and F 55 were recorded on three ASKANIA kinetheodolites. The analysis of the aerodynamic forces was carried out by Dr. G. BRAUN and were compared with wind tunnel tests. The position of the

† Supplementary Paper which was not read at the AGARD Seminar.

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controls, and the orders to change them, were produced by high frequency transmitters.



*Fig. 1. Research missile F 25 "Feuerlilie" of the Luftfahrtforschungsanstalt Braunschweig*

It was hoped to carry out tests at supersonic speeds, but no reports have been issued since the research programme was interrupted by the end of the war.

Because it was also necessary to give the project a direct military importance, the "FEUERLILIE" had to be planned as an anti-aircraft missile. Because of this different descriptions of the missile have been published based on a tactical "FEUERLILIE" although really only research problems were solved.

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